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STUDY OF A SOLAR SONDE IN THE ECLIPTIC WITH AN APPROACH TO
THE SUN TO WITHIN 0.26AU

D.E. Koelle et al.

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Study of a Solar Sonde in the
Ecliptic with an Approach to
the Sun to within 0.26AU

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1. PRESENTATION OF THE PROBLEM

According to the specification, the study shall investigate the feasibility, from the point of view of space technology, of a solar-sonde mission with an approach to the sun of closer than 0.3 AU.

This mission stands in the fore in discussions concerning German-American collaboration in the exploration of interplanetary space. It could represent a direct supplement to the NASA program, in which such a mission has so far not been incorporated.

In this field of research, there have so far been only the PIONEER-6 series of space sondes, which serve for the exploration of space between 1.2 AU and 0.7 AU.

In the case of the mission dealt with here, the exploration of space shall be extended to within 0.26 AU of the sun, and the influence of the sun in this region shall be determined. This, however, requires a sonde with orientation to the sun (in contrast to the relatively simple PIONEER device which is spin-stabilized). Furthermore, the considerably closer approach to the sun involves certain problems of temperature regulation and energy supply, which increase the weight and technical in-

put (the American PIONEER sondes possess a mass of 64 kg -- with 16 kg of scientific instrumentation - and their development cost 37.4 million dollars).

The "minimum instrumentation" preliminarily determined by the DKfW special "solar sonde" committee incorporates the following five experiments:

- (1) Measurement of the solar plasma present in interplanetary space (electrons, protons, heavy nuclei) over a larger energy spectrum.

Mass spectrometer: weight approximately 10 kg, energy requirement 5 w; information flow 10 bits/s.

Experimenter: Dr. Pinkau, MPI Garching

- (2) Measurement of the interplanetary magnetic field by means of three-component measurement.

3 Foerster sondes: 3.5 kg, energy requirement 5 w; information flow 1.7 bits/s.

Experimenters: Dr. Siemann and F.M. Neubauer, Institute of Geophysics and Meteorology of the Institute of Technology Braunschweig.

- (3) Measurement of cosmic radiation by counting electrons, protons and high-energy particles (three-component measurement)

3 end-window counter tubes and

4 disk counter tubes

or

telescope consisting of three semiconductor detectors.

In both cases 2.5 kg mass, 0.5 w energy requirement, information flow 0.5 bits/s.

- (4) Measurement of the distribution of the finest dust and free electrons in space by means of two-color photometry of the zodiacal light and the F corona (measurement at 3000 Å and 5500Å).

Photometer: 4 kg, energy requirement 2 w, information flow 0.1 bits/s

Experimenter: prof. Dr. Elsaesser, Landessternwarte (National Observatory) Heidelberg-Koenigsstuhl.

- (5) Measurement of micrometeoroids in space through registration of the impact pulses according to direction and magnitude (particles down to 10^{-10} g).

Mass spectrometer: 1 kg, energy requirement 2w, information flow 0.4 to 40 bits/s.

Experimenter: Prof. Dr. K. Sitte, MPI Heidelberg

The 5 experiments require a mass of 21 kg.

Two other experiments can be carried out without special instrumentation:

- (6) Investigation of the absorption and scattering of radio waves near the sun (K corona) through evaluation of the radio signals of the sonde.
- (7) The improvement of astronomic constants, such as the mass of the earth and the distance from the earth to the sun (astronomical unit), as well as the mass of Mercury.

The above-mentioned measurement tasks may, among other things, substantially contribute to the clarification of the following scientific problems:

- (I) The structure of the solar plasma currents; determination of the outermost limit of the (rigid) plasma rotation and thus of the sun's atmosphere (experiment 1 and 2).
- (II) The mechanism of the formation of proton flares.
(Experiments 1, 3, & 3).

- (III) The origin of solar cosmic radiation; screening off of the galactic cosmic radiation by sun or corona (exp. 3).
- (IV) The composition and microstructure of zodiacal light and of the F corona, the inner limit of cosmic dust in the solar system (experiments 4,5)
- (V) The mass and speed distribution of sporadic and stream meteoroids in the innermost region of the solar system (experiment 5).
- (VI) K-corona structure and reactions in the sun entailing phenomena in the corona (exp. 1,3,6).
- (VII) Solar-terrestrial relations (exp. 1,3,6).
- (VIII) More accurate values of important astronomic constants (experiment 7).

A complete solution of the above-mentioned problems in a first mission is at best possible in case of the limits of plasma rotation (I) and cosmic dust (IV); however, aside from their direct scientific value, all the results to be expected are of importance for the planning of further missions in the vicinity of the sun.

2. FUNDAMENTALS

2.1 Discussion of the Scientific Useful Load

2.11 Possible Experiments Concerning Sun Exploration

Table 2-I represents a general survey of the measurement tasks and possibilities of a solar mission.

Seven essential scopes of tasks with up to four (or more) measurement methods can be defined, aside from numerous combination possibilities.

From these, the DKfW special solar sonde committee has as a minimum mission selected 5 experiments, which chiefly serve for the exploration of space up to minimum distance from the sun.

Table 2-I shows that extension to the direct exploration of the sun (tasks 6 and 7) requires either a relatively large measuring instrument (coronaphotometer) and/or a close approach to the sun (x-ray spectrometer 0.1 to 0.2 AU). This complicates the mission and requires a relatively involved space-flight instrument, i.e. a "2nd-generation" solar sonde.

In Table 2-I the DKfW experiments of the minimum mission are marked by black dots. No final selection of the measurement instruments for cosmic radiation has yet been made.

2.11 Value and Extent of the Results as Function of the Distance of the Sun

As one of the fundamentals of the study, the extent to which an approach to the sun -- at the expense of the useful load -- is meaningful remains to be clarified.

In order to do this, it is necessary to obtain certain information concerning the value of the measurements to be expected as a function of the distance of the sun. This was attempted in TN-P6B-31/66, and the result is represented in Fig. 2-1 for the DKfW "minimum package" for the extended problem, and for a "2nd generation" solar sonde.

According to this, an approach to the sun of 0.26 AU is sufficient for the minimum package in order to obtain an optimum yield of results.

On the other hand, in the case of expanded experimental package, it seems necessary to come as close to the sun as 0.12AU. For

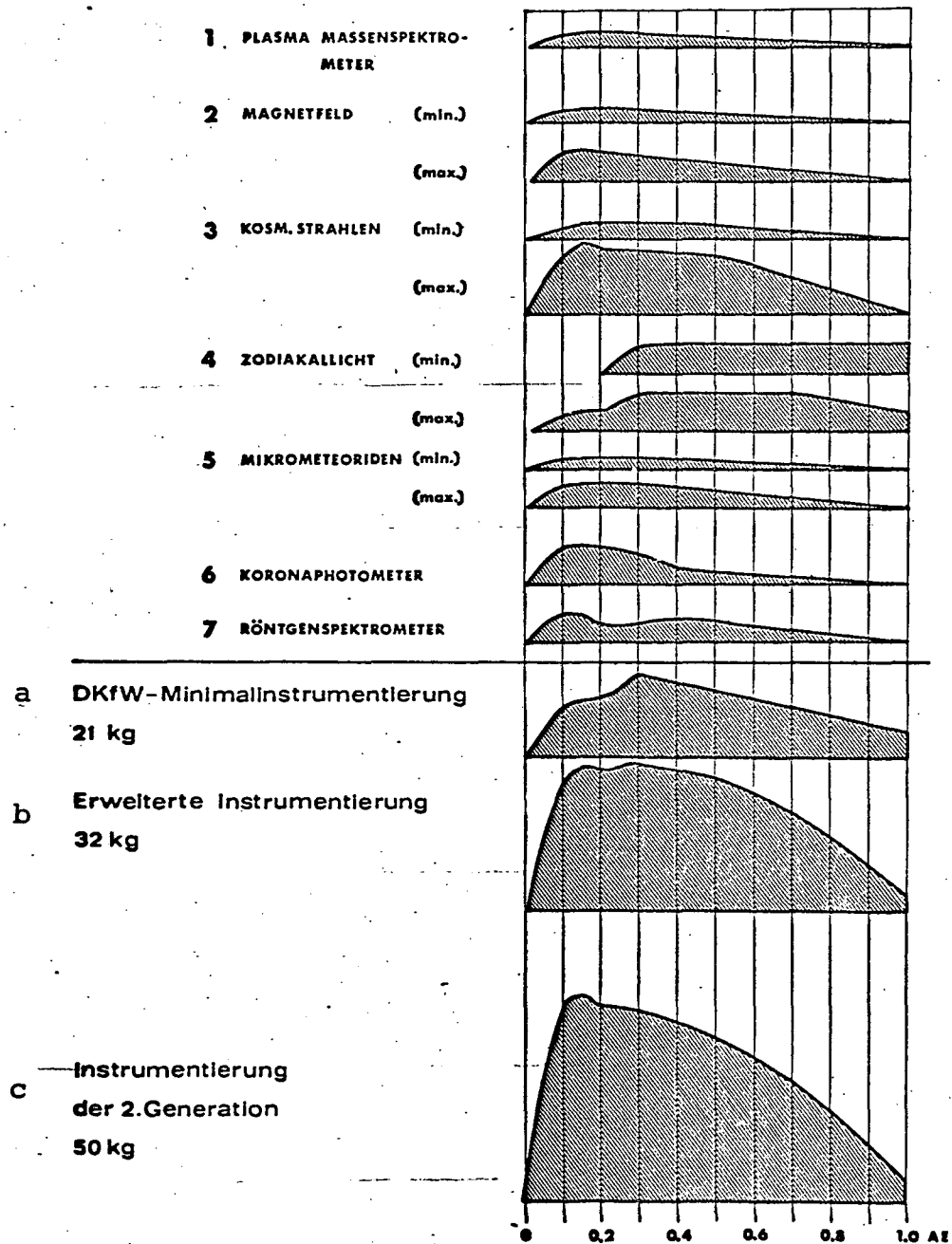
this reason, this mission was designated as a 2nd-generation solar sonde, since it will require a sonde with a mass of more than 400 kg.

The individual diagrams in Fig. 2-1 concerning the expected value of the experimental results as a function of the distance of the sun were established empirically and are only meant to serve as a demonstration.

Table 2-I: Survey of Possible Experiments Concerning a Solar-Sonde Mission

TASK	MEASUREMENT DEVICE	Weight kg	Capacity Watts	Bit rate bits/s	Storage bits
1 Interplanet. Plasma	(a) mass spectro- meter	10	5	10	1 000
2 Magnetic field	(a) 3 Foerster sondes	3,5	5	1,7	-
	(b) μ - Magneto- meter	3,5	5	2,7	-
3 Cosmic Radiation	(a) 3 end-window countertube	1,0	0,2	0,2	5 000
	(b) 4 Disk meter tubes	1,5	0,2	0,2	5 000
	(c) Semiconductor telescope	2,5	0,7	0,3	10 000
	(d) Cerenkov- telescope	4,0	1,5	0,3	10 000
4 Zodiacal Light	(a) 2-color spectrometer	4,0	2,0	0,1	-
	(b) Balmer- Experiment	4,0	2,0	0,1	-
	(c) Lyman- α - Experiment	3,0	3,0	1,0	-
5 Micrometeoroids	(a) M counter	1,0	2,0	0,4	-
	(b) Mass spec- trometer	1,0	2,0	10	40 000
	(c) Microphone counter	0,3	0,1	0,1	-
6 Corona	(a) Photometer	15	6	1	4 000
7 X-radiation	(a) X-ray spectrometer	3	10	1	10 000

FIG 2 - 01



Survey of the expected yield of the experimental results as a function of the approach to the sun.

1, plasma mass spectrometer, 2 magnetic field, 3 cosmic rays
4 zodiacal light, 5 micrometeoroids, 6 corona photometer, 7 X-ray spectrometer

a= DKfW minimum instrumentation, 21 kg, b=pending instrumentation 50 kg, c= 2nd generation instrumentation 50 kg.

2.13 Requirements of the experiments concerning Position Regulation

Basically, the sonde can be spin-stabilized (like Pioneer 6) or can be position-stabilized (like Mariner); decisive here is the instrumentation.

Out of the 5 experiments of the DKfW, the plasma detector and the zodiacal light experiment require orientation to the sun. In this case preference must be given a position-stabilized sonde, which generally facilitates the evaluation of the experiments and possesses certain technical advantages. The accuracy requirements are not so high as to involve special problems with regard to position regulation.

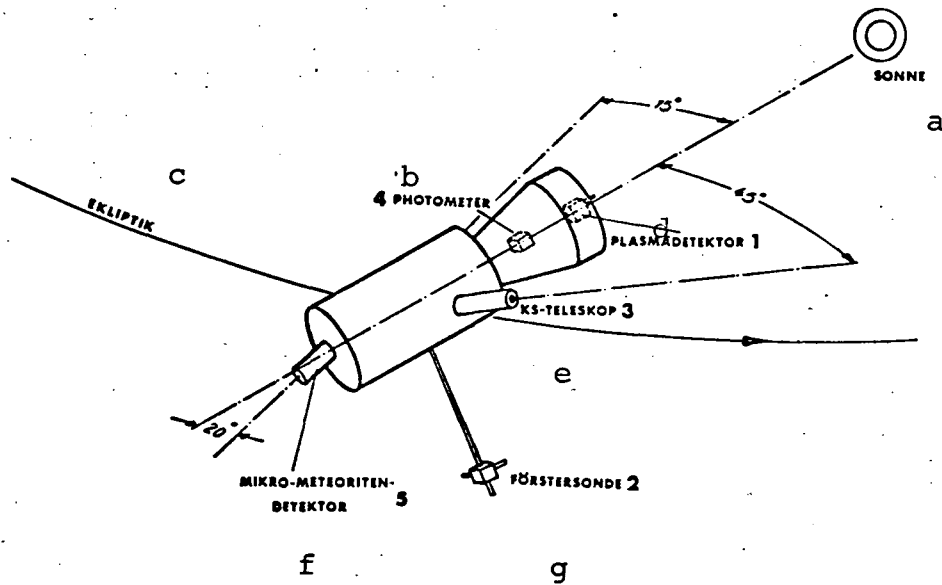
- (1) The plasma experiment permits a deviation from the direction of the sun by $\pm 20^\circ$ but requires the transmission of orientation within $\pm 1^\circ$.
- (2) The magnetic-field sondes do not require sun orientation, but transmission of the orientation within 4 arc minutes.
- (3) The counter tubes for the cosmic radiation must be either oriented to the sun at $\pm 10^\circ$ or to a point in approxi-

mately 45° elongation. The orientation must be transmitted within $\pm 1^\circ$.

- (4) The requirements of the two-color photometer (zodiacal light) are determined by the fact that the coelostats must not be hit by scattering light from the sonde and the K corona, i.e., at 0.2 AU distance from the sun from an area with a radius of 3.5° around the center of the sun. This means a position-regulation accuracy of $\pm 2^\circ$ and 15' measurement accuracy.
- (5) The micrometeroid experiment shall principally be oriented in the ecliptic perpendicularly to the flight-orbit plane and to the orbit vector. Deviations up to 40' are permissible, but the orientation must be transmitted within 1° .

This survey shows that the sonde must be oriented to the sun with a precision of $\pm 2^\circ$ whereas a navigation accuracy of 4 arc minutes is necessary.

FIG. 2 - 02



Geometric installation conditions of the
planned experiments.

- a = sun
- b = photometer
- c = ecliptic
- d = plasma detector
- e = KS detector
- f = micrometeorite detector
- g = Foester sonde

2.14 Installation conditions of the experiments

(1) PLASMA EXPERIMENT:

A mass spectrometer to be newly developed, which represents an improved version of the VELA spectrometer shall be used for this experiment.

From this, a mean diameter of the new device of approximately 18 cm and a construction length of approximately 25 cm can be estimated (including current supply).

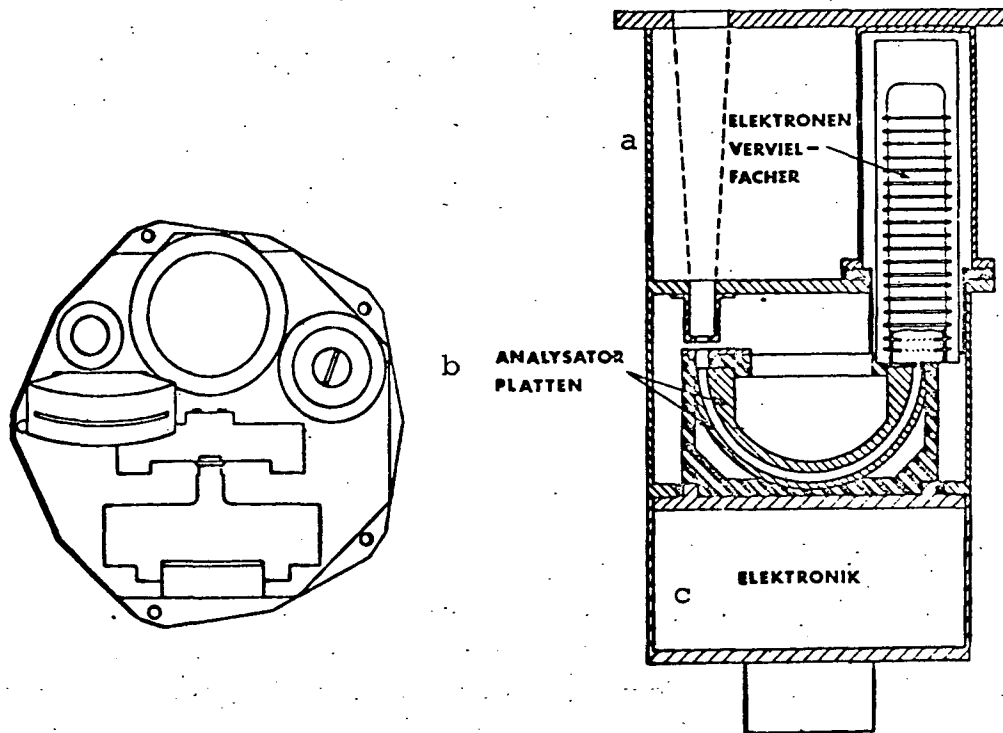
The device shall be sun-oriented and could be installed directly behind the heat shield.

Permissible temperature range -20° to $+80^{\circ}$ C.

(2) MAGNETIC-FIELD EXPERIMENT:

Three Foester sondes (for each of the three axes of the device) are incorporated here as measurement devices. Each of them possesses a diameter of 70mm and a height of 100 m. The permissible temperature range of the sensors lies between -55° C and 80° C, for the electronic equipment it is between -40° and 70° C.

FIG. 2 - 03



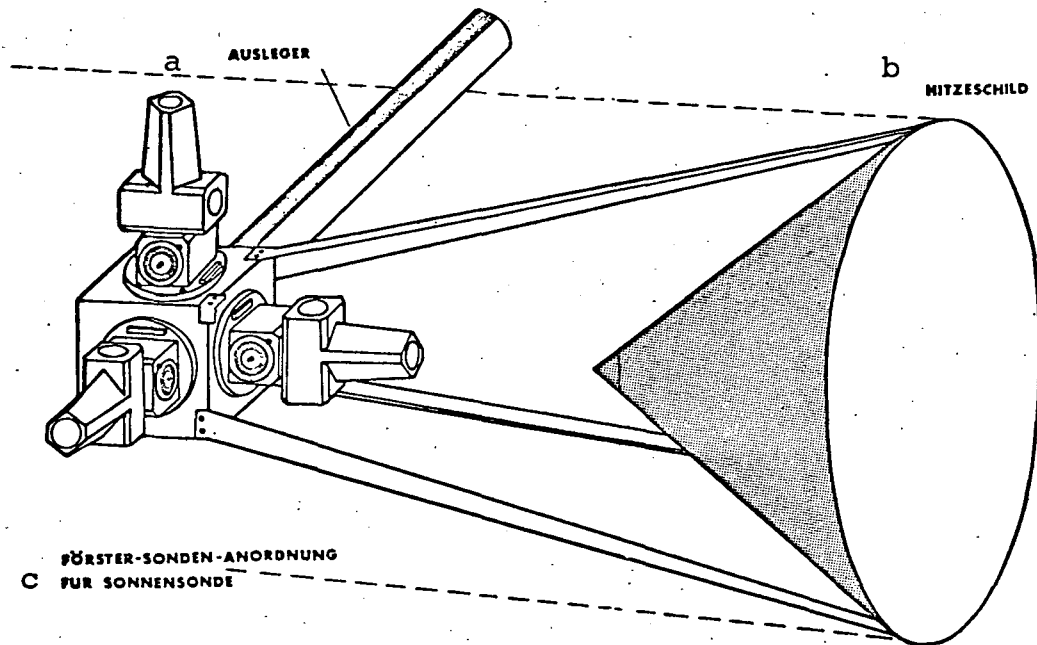
Structural principle of the mass spectrometer

a = analyzer plates

b = electron multiplier

c = electronic equipment

FIG. 2 - 04



Measurement of the interplanetary
magnetic fields through three perpen-
dicularly opposed Foester sondes.

a = arm

b = heat shield

c = Foester-sondes arrangement
for solar sonde

(3) COSMIC RAY MEASUREMENT

The counter-tube arrangement has a mass of approximately 2.5 kg, and the volume of the measurement arrangement amounts to approximately 2500 cm³. The geometry may be varied within certain limits.

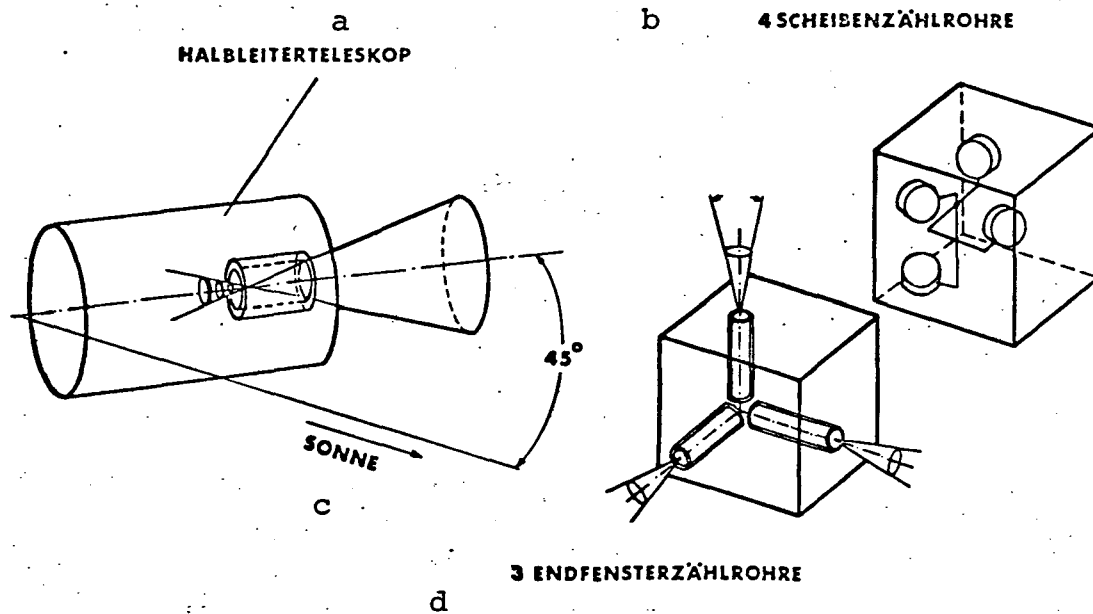
No experience is yet on hand with regard to the long-period behavior of semiconductor detectors at higher temperatures, therefore no information can be given concerning the permissible temperature range. Lower temperatures must be preferred because of the influence of temperature upon the inverse current and noise of the counters.

(4) TWO-COLOR PHOTOMETER:

The configuration of the entire sonde is to a considerable extent determined by the two-color photometer for registration of the stray light of the interplanetary medium (zodiacal light).

Since no such experiment has yet been flown, it is, in addition, of particular significance.

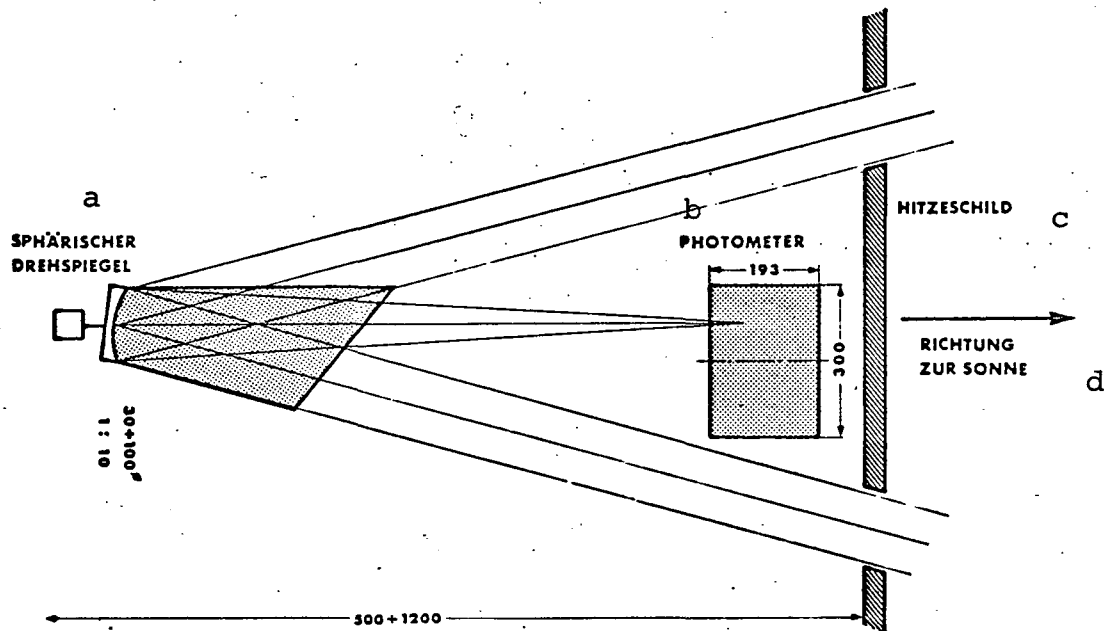
FIG 2 - 05



Schematic ray telescope with semiconductor detectors (left) and the counter-tube combination (right)

- a = semiconductor telescope
- b = disk counter tubes
- c = sun
- d = end-window counter tubes

FIG. 2 - 06



Installation geometry of the
two-color photometer

a = spherical revolving mirror

b = photometer

c = heat shield

d = direction to the sun

Since an area at a distance of approximately $15^{\circ} \pm 2^{\circ}$ to the sun shall be scanned, but on the other hand no direct sun light may incide (also no stray light) a conical tubus must be incorporated.

The light intensities and geometric relationships then yield constructions lengths of between 500 and 1200 mm from the front plate to the mirror (fig. 2-06).

The mirror shall be turned stepwise 45° every 60 sec. so that 8 measurement areas around the sun can be scanned within 8 min (with one filter). Since two filters are provided, one measurement series consequently lasts 16 min.

The photomultiplier must be particularly protected against heat, the temperature should not exceed 20° C.

Since the function of the photomultiplier can be influenced by outer magnetic fields, which bring about a deformation of the electron orbits and thus a change of the focusing, screening must, if required, be provided for.

(5) MICROMETEOROID EXPERIMENT

By means of a new measurement method, particles as small as 10^{-16} g shall be measured here, whereas up to now only particles to approximately 10^{-16} g could be measured with the aid of mechanical sensors.

The measurement arrangement consists of a micrometeoroid catcher (with a surface of approximately 10 cm^2) acceleration grids, and a detector for electrons and ions. The dimensions of the cylindrical arrangement are approximately $10 \text{ cm} \times 20 \text{ cm}$.

The catching surface shall be oriented in the ecliptic perpendicularly to the direction of the flight path; deviations of $\pm 30^\circ$ from this direction are, however, permissible.

The temperature of the measurement arrangement should not exceed 80° C , room temperature would be desirable.

It is planned to test this new measurement principle during high-altitude rocket launching in 1967.

FIG 2 - 07

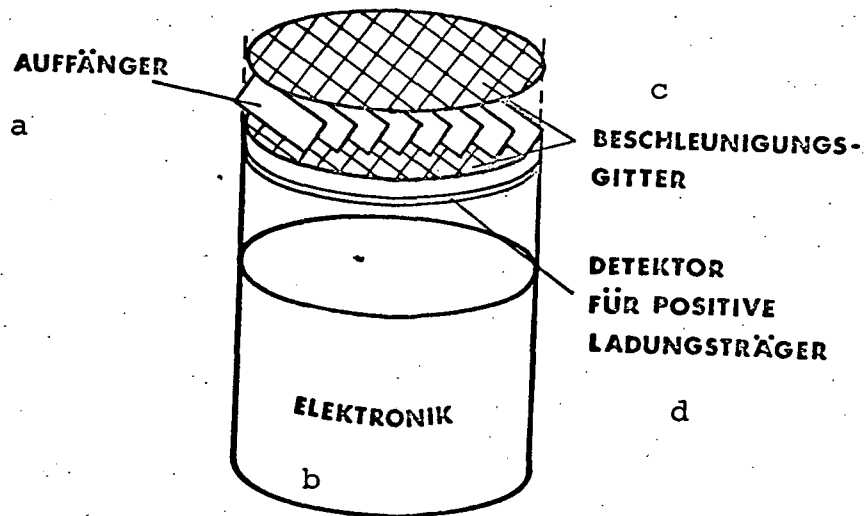


Diagram of the meteoroid detector according
to Prof. Sitte.

a = catcher

b = electronic equipment

c = acceleration grid

d = detector for positive charge
carriers.

2.2 Launching and Thrust requirements (Speed requirements)

The launching of a solar sonde requires a relatively high energy expenditure, comparable with that of sondes to the outer planets (Jupiter, Saturn, Neptun), or even more if a closer approach to the sun is desired than 0.2 AU (1 astronomical unit = 1.498×10^8 km (FIG. 2-08) .

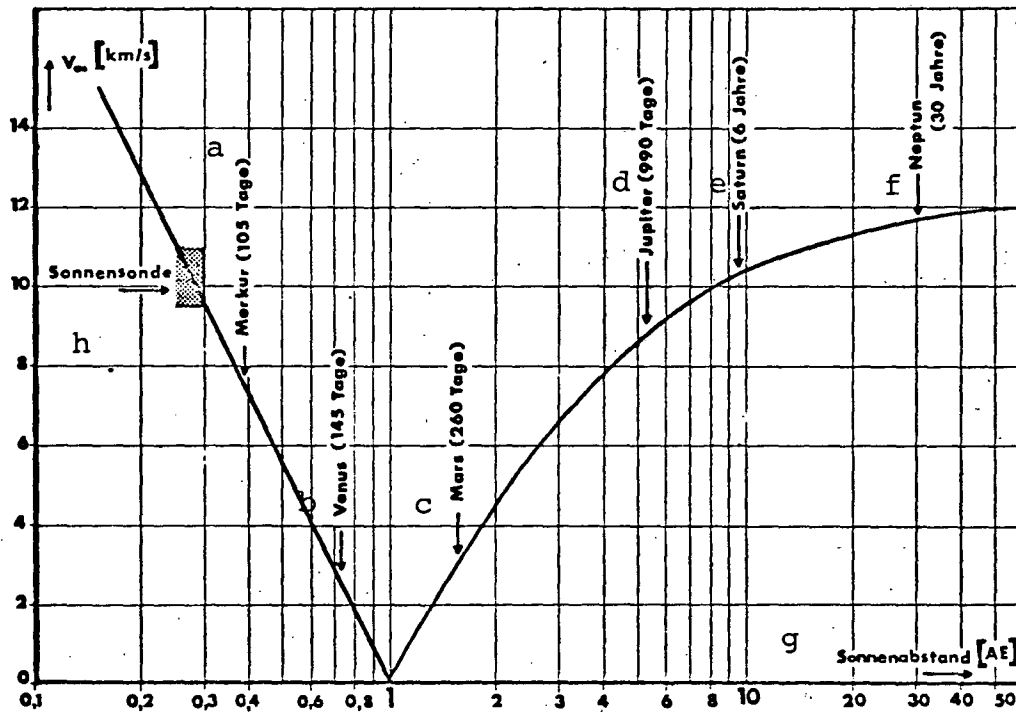
Figure 2-09 shows a survey of the necessary thrust requirement with 1.6 km/s losses for ecliptical and ex-ecliptical solar sondes.

Here it is interesting that in the case of ex-ecliptical sondes, there is each time an optimal perihelion distance, according to the heliocentric width.

The exact required thrust of a solar sonde is a function of the following parameters:

1. the orbit profile up to injection (according to carrier rocket)
2. the desired perihelion (point nearest to sun)
3. the time of launching (or, as the case may be, the exact distance from sun)
4. the propulsion system of the sonde (magnitude of thrust and engine lifetime).

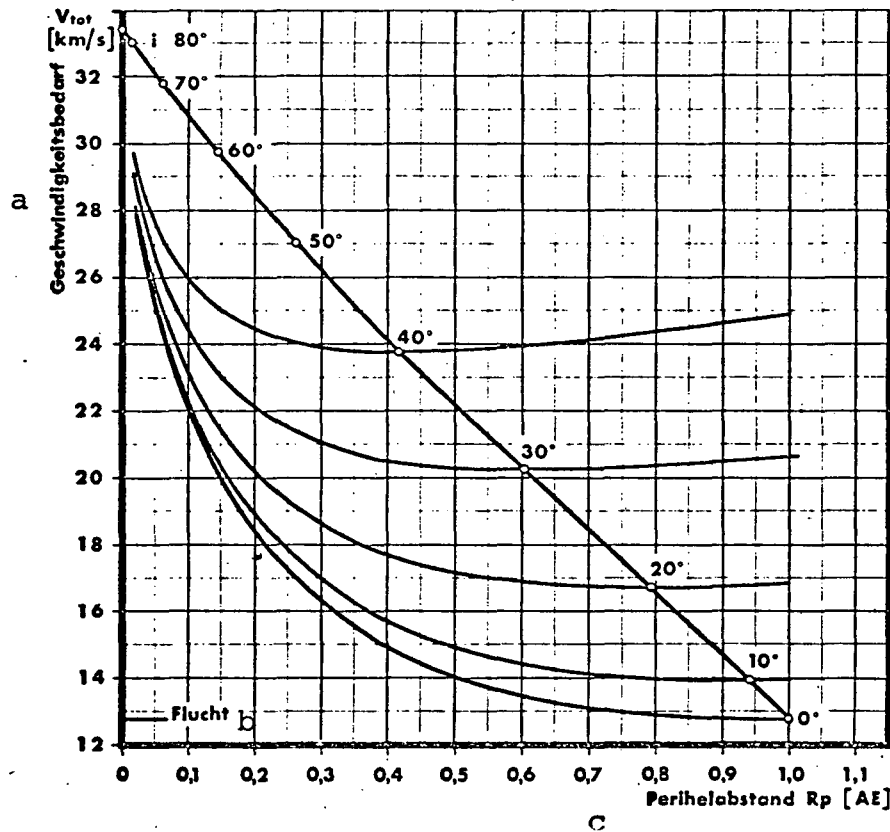
FIG 2 - 08



Hyperbolic excess velocities required for interplanetary missions (Hohmann-transition).

- a = Mercury (105 days)
- b = Venus (145 days)
- c = Mars (260 days)
- d = Jupiter (990 days)
- e = Saturn (6 years)
- f = Neptune (30 years)
- g = distance from sun (AU)
- h = solar sonde

FIG 2 - 09



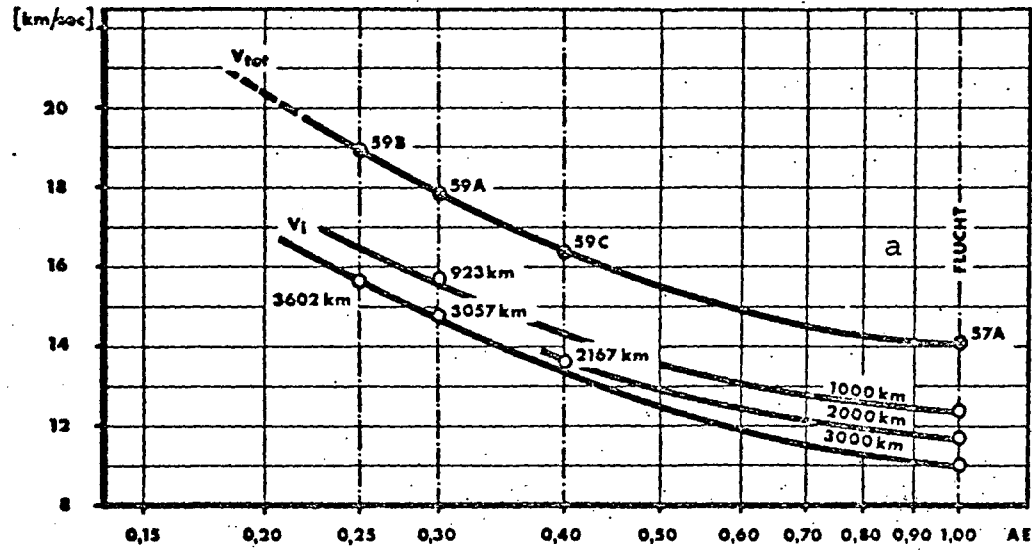
Total velocity requirement as function of the perihelion distance

a = velocity requirement

b = escape

c = perihelion distance

FIG 2 - 10



Thrust required for the injection of
a solar sonde from Cape Kennedy (ATLAS-
CENTAUR + highly energetic propulsion
unit)

a = escape

This exact required thrust can generally only be determined by orbit calculations. FIG. 2-10 shows the results of such calculations with the ATLAS-CENTAUR carrier rocket + highly energetic propulsion unit for the sonde. The injection or propellant-cutoff velocity decreases with increasing propellant-cutoff altitude, whereas the losses increase (differences between V_{tot} and V_i).

FIG. 2-11 illustrates the influence of the starting time; the variation of the required thrust is due to the excentricity of the earth orbit. It is therefor advantageous to launch the sonde in the summer (June/July). The minimum propulsion requirement occurs on 2/3 July (of each year), the launching time in the case of direct injection being 5.54 and 17.52 (on July 2).

A deviation from the planned start time requires the addition of a parking orbit. This launching delay may not become too great because of the changeable position of the earth, since otherwise the deviations of the injection parameter exceed a permissible measure.

The ascent into the parking orbit then no longer takes place in a plane, i.e. the circular-orbit altitude, in contrast to the direct ascent, is attained at a different point, which is, with reference to the original one, displaced by a certain angle. The

duration of the free-flight phase in the parking orbit is then equal to the time difference between a complete rotation and the duration in which the angle difference is travelled. The addition of a parking orbit is, however, connected with a loss of useful load (due to the required additional stage weights for free flight, pre-acceleration and re-ignition); moreover, dependability is thereby somewhat reduced.

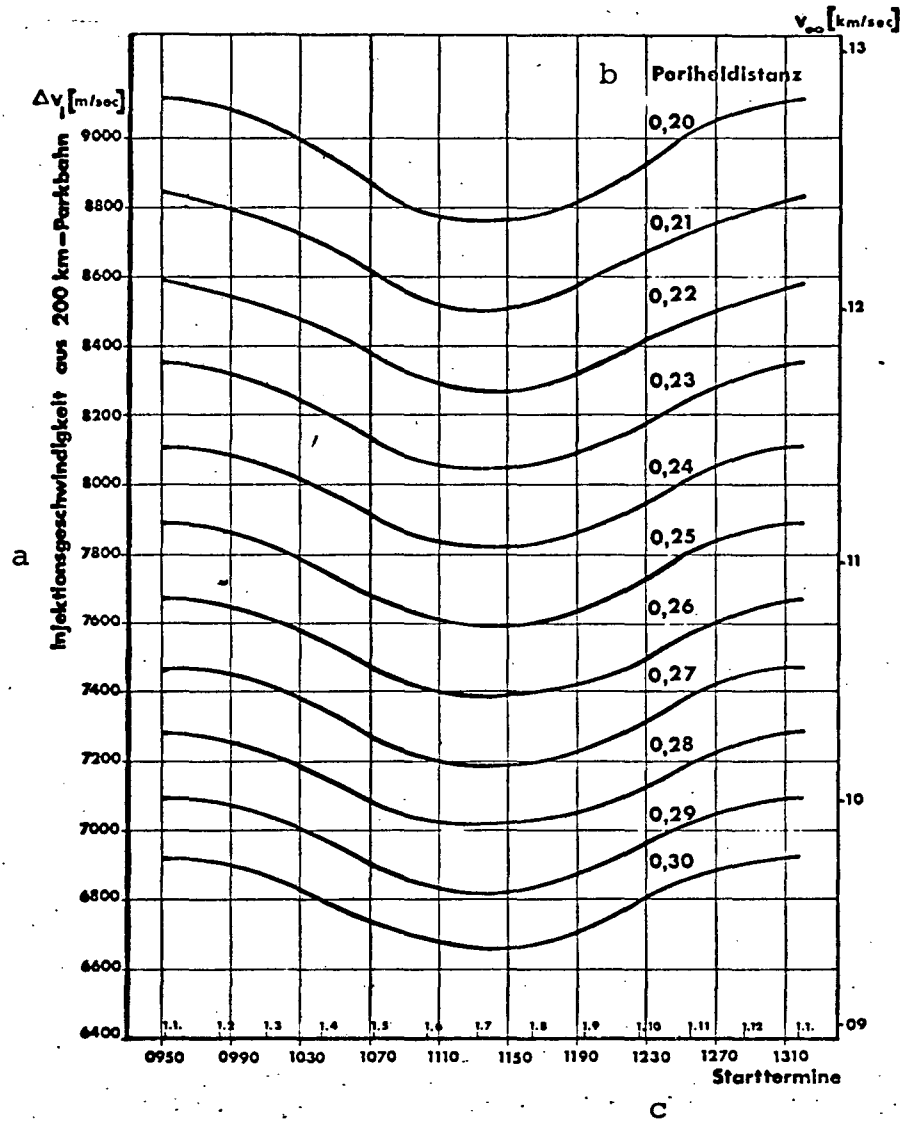
2.3 Construction of a solar sonde

The project of a solar sonde is characterized by problems of heat economy, energy supply and data transfer. In the case of an approach to less than 0.26 EU, the corresponding problems increase considerably. Up to this perihelion distance, solar-battery outriggers are, for instance, still possible.

On the other hand, no course-correction system is necessary and the position regulation becomes relatively simple. A weight estimate of the sonde must be made for the investigation of a suitable carrier system.

Static data of similar space sondes, as well as the weights and the energy requirement of the three experimental "packages" may here serve as an initial basis.

FIG 2 - 11

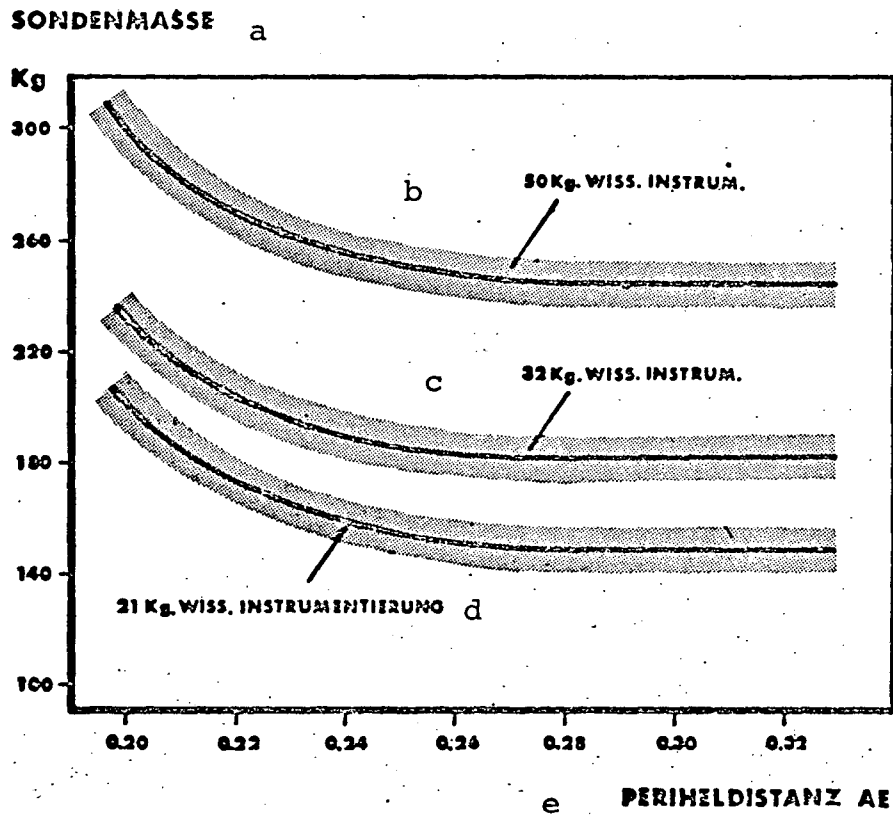


Influence of the launching time upon the velocity requirement

a = injection velocity from 200-km parking orbit

b = perihelion distance c = launching times

FIG 2 - 12



Required sonde mass as function of the
perihelion

a = sonde mass

b = 50 kg scientific instrumentation

c = 35 kg scientific instrumentation

d = 21 kg scientific instrumentation

e = perihelion distance AE

Detailed investigations, which are written down in TN-P6B-33/66 resulted in the curves represented in FIG 2-12.

The weights and weight ranges are in reference to a sonde with no considerable redundance and without flight past Venus. The flight past Venus would increase the weights by approximately 17%, since in addition a course-correction system and an expanded position-regulation system becomes necessary.

The individual weights are compiled in Table 2 - II; here, however, we are dealing only with rough estimates. The required electric capacities were compiled in the same manner. They amounted to 92 w for 21 kg and 155 w for 32 kg scientific instrumentation.

The cited figures correspond to the present-day state of technology, since for reasons of expense and dependability it makes little sense to incorporate extremely light-weight constructions.

TABLE 2-II - Estimate of the sonde weight as function of the instrument weight

M INST	21 kg	32 kg	50 kg
Computer + Sequencer	7	8	11
Antennas	4	4	4
Storage	4	6	4
Communication receiver + processing	5	5	5
Sender + data process.	10	13	15
Position regulation	18	20	25
Energy supply	23	28	33
Structure	40	45	55
Safety margin 15%			
TOTAL	152 kg	185 kg	240 kg

2.4 Possible Carrier Rockets

2.41 Carrier Rockets without auxiliary Stages

At the present time there does as yet exist no carrier rocket which could bring into orbit a solar sonde with a perihelion < 0.4 AU; however, at the time of the possible launching of a solar sonde (1972), SATURN V will be available (FIG 2-13).

With SATURN V, a sonde of up to 800 kg can be brought to 0.23 AU -- or two sondes can be launched simultaneously, which would no doubt be optimal from the point of view of reliability and of the scientific experiments.

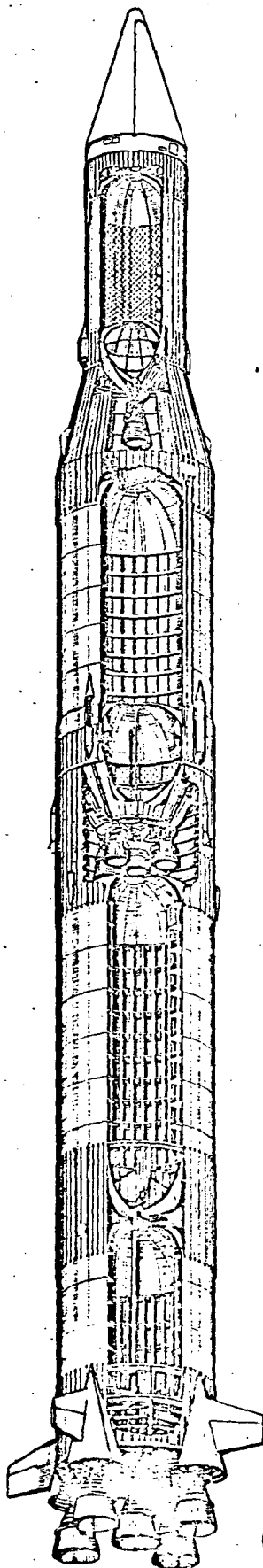
FIG. 2-14 shows the useful-load capacity of SATURN V. At the same time, the possibilities are plotted, if CENTAUR or a smaller stage or both are additionally employed: in this case a sonde with 400 kg mass could be brought up to 0.05 AU from the sun.

1. 42 Carrier rockets with existing auxiliary stages.

As it has already been mentioned in the preceding chapter, the useful-load capacity of a carrier rocket for interplanetary missions can be increased by means of additional stages.

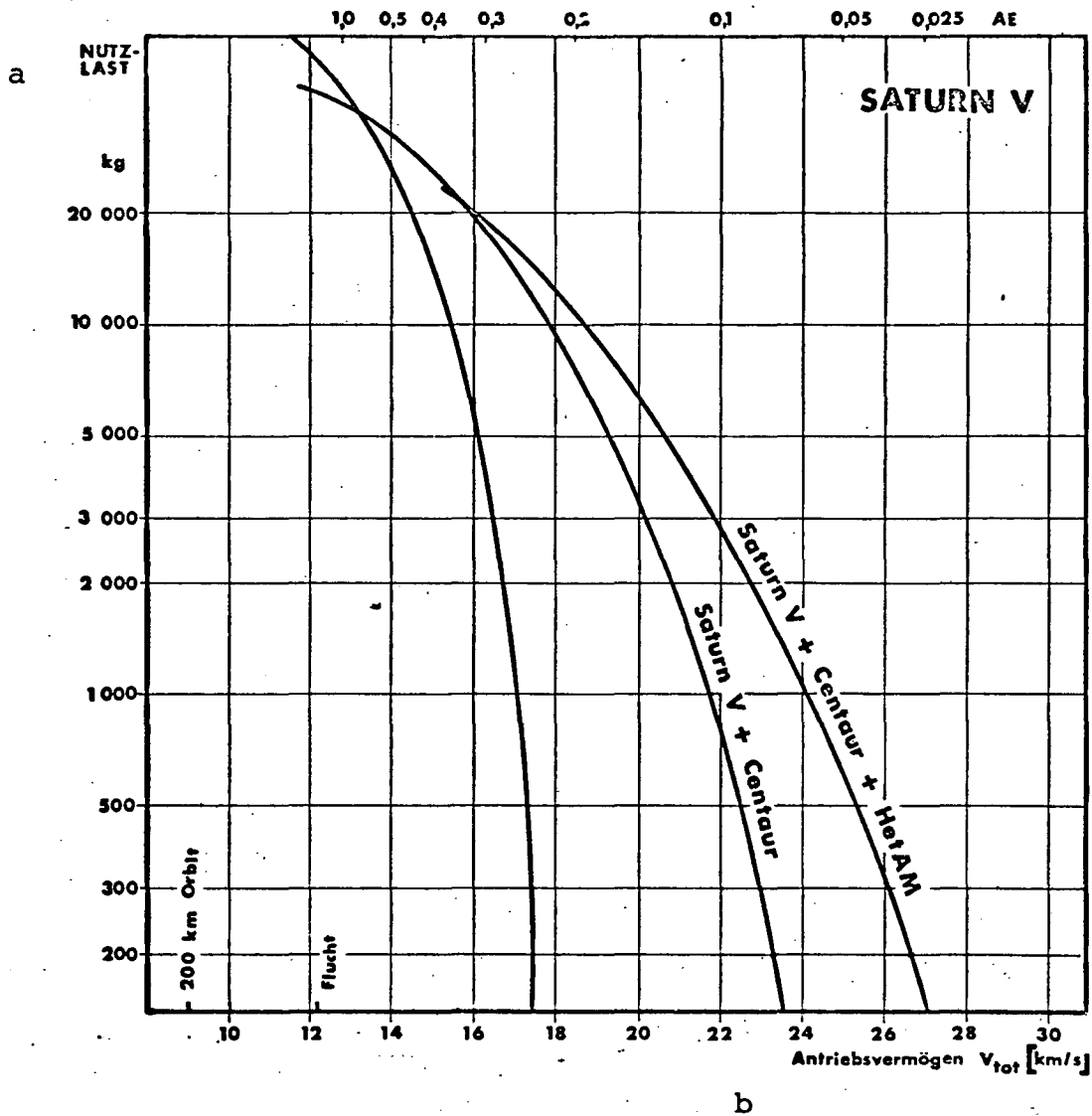
FIG 2-13

SATURN-V



Reproduced from
best available copy.

FIG 2 - 14



Useful-load capacity of SATURN V for
solar sondes (without or with additio-
nal highly energetic stages)

a = useful load

b = escape

c - gravity-free velocity at end of active
trajectory phase

The following typical representatives of three different types and dimensions of "kick stages" are basically possible for this purpose:

(A) BURNER II (Boeing)

Kick stage with thickol solid-fuel engine; total mass 770 kg
engine 625 kg, fuel 560 kg

Specific impulse: 280 sec (estimated)

Required system reliability 96%.

(B) ELDO-A-THIRD STAGE (Boelkow-ERNO)

Fuels: Aerozin 50/ N_2O_4 , specific impulse 280 sec

Net weight 750 kg, fuel mass 2800 kg

Diameter 2.0 m

(C) CENTAUR (General Dynamics)

Fuels: hydrogen/oxygen

Net weight 1900 kg, fuel mass 13,600 kg

Diameter 3.05 m

Hereby the following new carrier rocket versions are obtained:

- (1) ATLAS-AGENA-BURNER II
- (2) ATLAS-CENTAUR-BURNER II
- (3) ATLAS-CENTAUR-ELDO A - III
- (4) SATURN I B - BURNER II
- (5) SATURN I B - ELDO A -III
- (6) SATURN I B - ELDO A - III
- (7) SATURN I B - CENTAUR

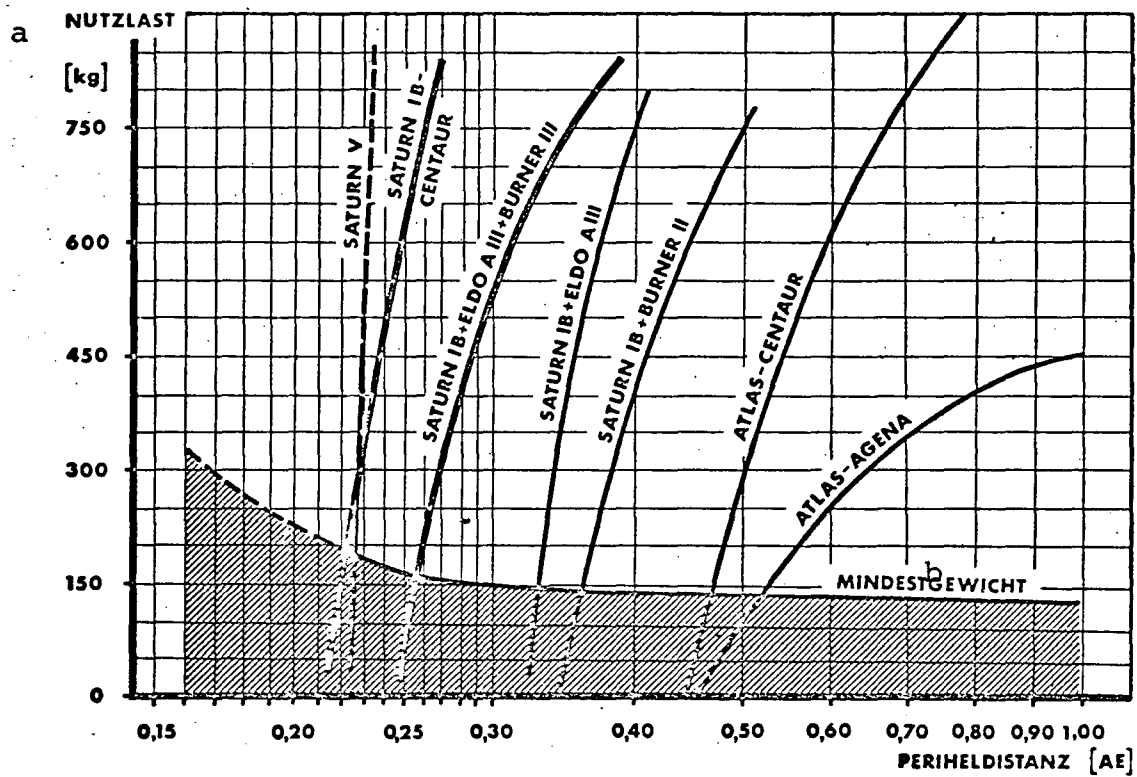
If the possible useful load of the seven above-mentioned carrier-rocket combinations is plotted against the perihelion distance, FIG 2-15 is obtained.

With 0.3 AU as a limit, only two carrier systems -- both of which use SATURN I B as basis -- remain in addition to the already mentioned SATURN V (FIG 2-16).

- (1) SATURN I B - ELDO A III - BURNER II:

This combination is technically feasible and would have the advantage of including in the program the German IIIrd stage of ELDO-A; there would, however, be a certain expense in order to integrate this step into an American carrier rocket. (Problem of the additionally required ground installations at Cape Kennedy;). It will presumably be worth while only if several programs with this configuration are planned.

FIG 2 - 15



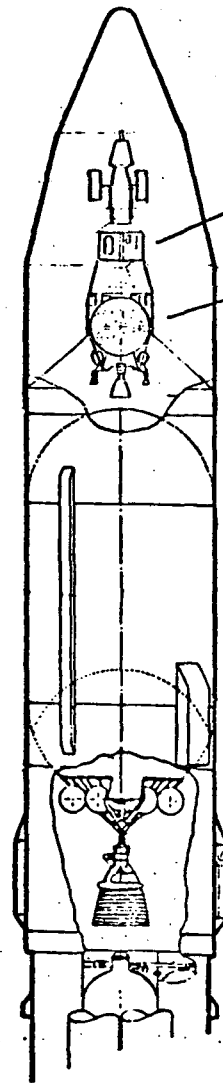
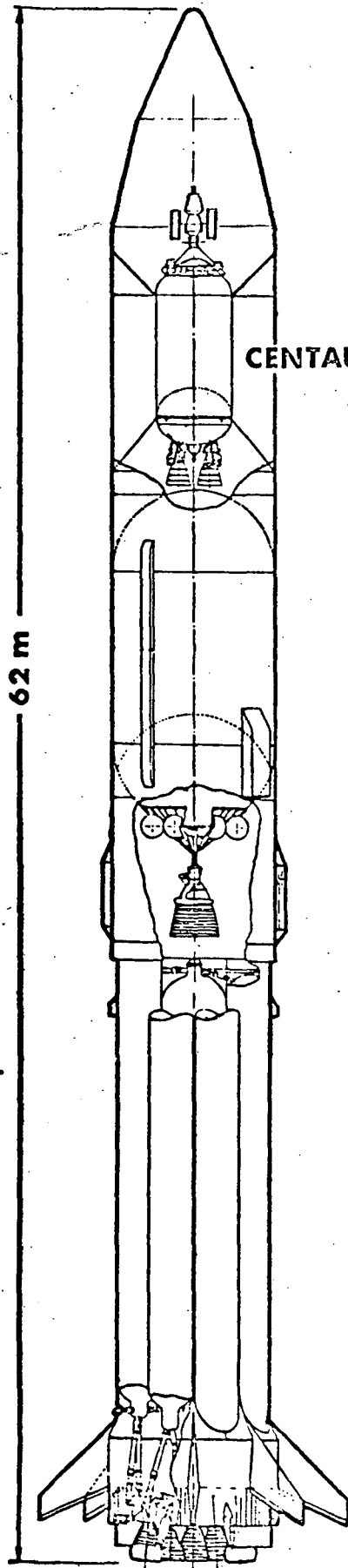
The useful capacity of various
carrier rockets with supplementary
final stages for a solar sonde

a = useful load

b = minimum weight

c = perihelion distance

FIG 2 - 16



BURNER II

a

III. STUFE

b

a= Burner II
b= III stage
c= SATURN IB with
supplementary stages
Launching weight
565 t

SATURN IB

MIT ZUSÄTZLICHEN
STUFEN

c

STARTGEWICHT 565 t

(2) SATURN I B - CENTAUR

The most reasonable and the most efficient solution would be the combination SATURN I B - CENTAUR, since it could be employed for many interplanetary missions.

This carrier system was conceived by NASA already in 1964; however, the planned development was cancelled due to financial considerations. Instead, SATURN V is now scheduled for the "Voyager-Mars" program (with the parallel launching of two space flight devices).

If the development of SATURN IB - CENTAUR should be resumed, this would be the preferred possibility for the start of a solar sonde; unfortunately however, there are as yet no signs of this.

2.43 Possibilities of an additional sonde propulsion unit (in connection with ATLAS-CENTAUR).

It is basically possible to equip the sonde with its own propulsion unit. This can be:

- (A) a solid fuel engine (or medium-energetic propulsion system)
- (B) a highly energetic propulsion unit (with liquid fuels, for example hydrogen/fluorine),

(C) a radioisotope propulsion unit (heating of hydrogen in a nuclide generator), and

(D) an electric propulsion unit with nuclear or solar energy supply.

In each of the four cases, ATLAS-CENTAUR can be employed as a carrier system (at least for the minimum useful load of 21 kg scientific instrumentation), and due to cost considerations this will probably be done.

Detailed investigations, set forth in chapter 3, which follows, led to the following results with respect to performance.

(A) State optimization yields for a solid-fuel engine (or for a corresponding medium energetic propulsion unit) an optimum fuel mass of approximately 2,000 kg. However, in case of the employment of a fuel factor of 0.90, and 120 kg of mass for position regulation purposes and additional structure, insufficient performances (only 100 kg 0.3 EU or 150 0.32 EU) are obtained, even at 295 sec specific impulse.

These data are oriented to the BURNER II final stage,

a new development of the Boeing firm, which employs the Surveyor solid-fuel engine, considered to be one of the most modern and most efficient solid-fuel engines in the USA. (Housing weight 65 kg with 560 kg fuel; 145 additional weight for suspension, position regulation, transponder, etc.).

(B) An optimal highly energetic propulsion unit will have approximately 1.6 t fuel, namely fluorine/hydrogen. This, first of all, in order to attain the highest possible performance, and secondly in order sensibly to employ the H_2/F_2 engine with approximately 550 kp vacuum thrust, which is presumably to be developed within the national program as of 1967. The estimated performance is approximately 440 sec.

(C) If a radionuclide generator is employed for current supply, it becomes obvious to use the excess energy of heat for the heating of hydrogen. This system has received the designation RAMSES= Radioisotopen-Antriebs-Modul fuer Sonden zur Erforschung des Sonnensystems (Radio isotope propulsion unit for sondes for exploration of the solar system). Here a thrust of approximately 1 N at a specific impulse of 750 sec is realizable.

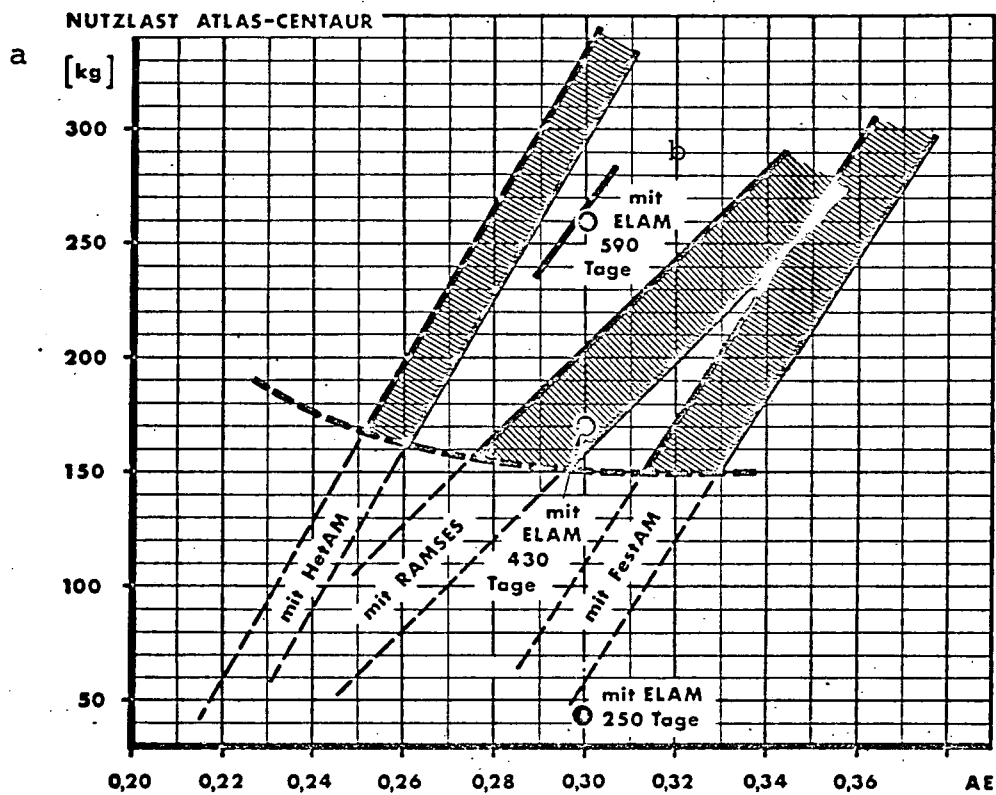
- (D) The fourth, and by far the most progressive solution is the employment of electric engines in connection with a reactor or, in addition, solar-battery surfaces.

Calculations were carried out for azimuthal thrust, since the realization of a constantly optimal thrust vector involves difficulties, mathematically as well as technically. We are here dealing with a package of approximately 30 electric engines, the total thrust of which is approximately 1.6 N and which requires 60 kw electric capacity. Moreover, the duration of the flight will be very considerable (600 days in comparison to approximately 90 days in the case of chemical drive), and the specific weights of reactor, converter and electric engines are at the present time still relatively uncertain, so that the useful-load values are also uncertain.

FIG. 2-17 shows a comparison of performance of the four propulsion systems; here an ATLAS-CENTAUR rocket is used as the basis.

Due to the necessary masses for the reactor, etc., electric propulsion requires a medium-altitude orbit, for example 500 km, where the useful load or initial mass of the sonde + El AM (Elektrisches Antriebsmodul -- electric propulsion unit is about 4,000 kg.

FIG 2 - 17



mit = with
Tage = days

a = useful load ATLAS-CENTAUR

Comparison of performance of different propulsion units for a solar sonde (launching with ATLAS-CENTAUR).

The solid-fuel unit and the high-energetic chemical propulsion system are already ignited below escape velocity, the initial mass in this case is approx. 2,200 kg.

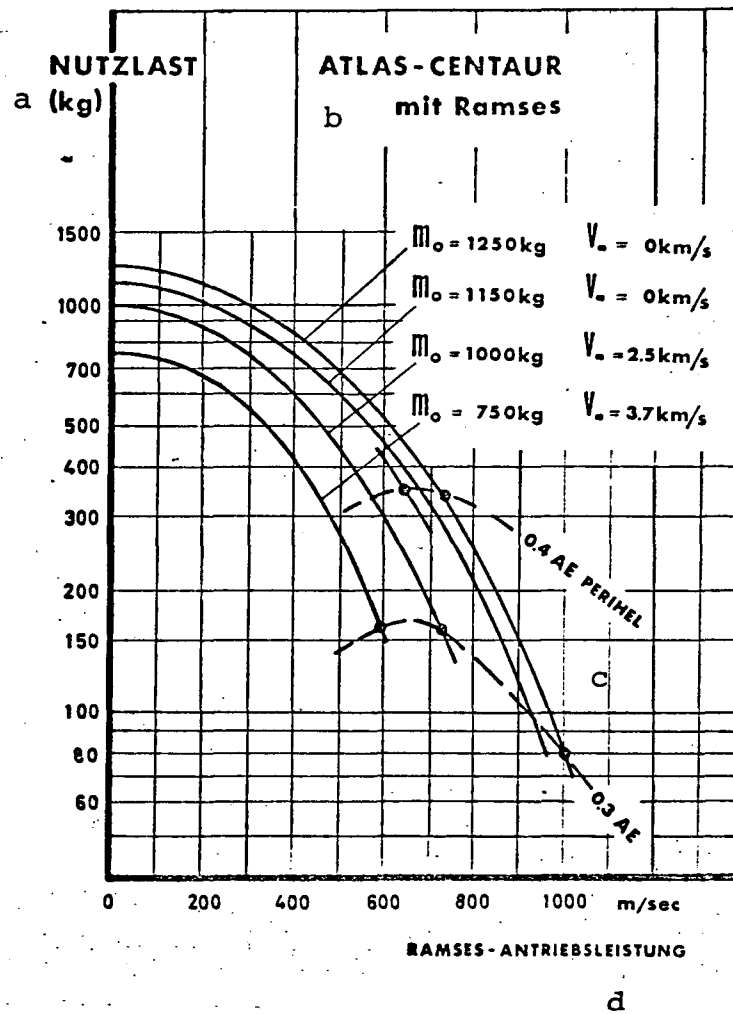
On the other hand, radioisotope propulsion requires an initial velocity, which is already considerably above the escape velocity, so that only a residual necessary thrust of 600 to 700 m/sec remains.

The initial mass of sonde + RAMSES is in this case approx. 850 kg. (FIG. 2-18).

With regard to performance analysis, the three other propulsion systems are feasible for a solar sonde with 0.3 AU perihelion except the solid-fuel engine, so that other viewpoints such as expense and time input, safety, and reliability must be decisive.

This will be discussed in the following chapter.

FIG 2 - 18



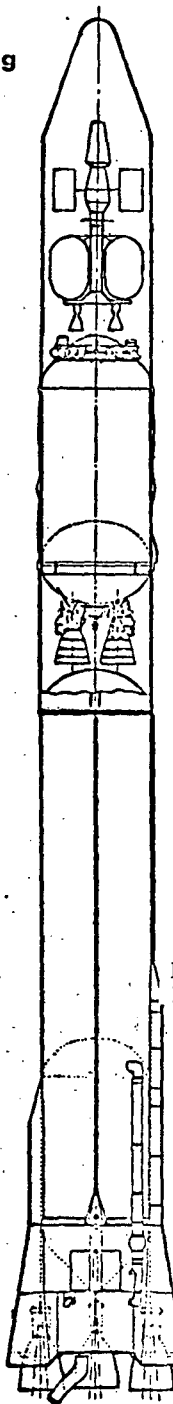
Useful load and initial load for RAMSES
when employed for a solar sonde.

a = useful load (kg)

b - ATLAS-CENTAUR with RAMSES

c - RAMSES-propulsion capacity

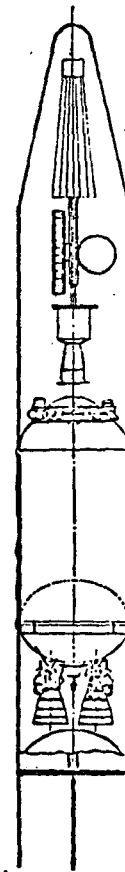
FIG 2 - 19
with
HetAM
 $M_0 = 2100$ kg



with
RAMSES
 $M_0 = 850$ kg



with
ELAM
 $M_0 = 4000$ kg



ATLAS (SLV-3X)-CENTAUR

Principal data for orbit calculations

Launching (without useful load)	M_0	=	147 000	kg
Fuel mass Ist stage Stufe	M_8	=	110 020	kg
Fuel mass IIInd stage	M_8	=	11 000	kg
Fuel mass IIIrd stage	M_8	=	13 580	kg
Specific impulse Ist stage	I_{sp}	=	150/295	sec
Specific impulse IIInd stage	I_{sp}	=	328	sec
Specific impulse IIIrd stage	I_{sp}	=	443	sec
Launching thrust Ist stage	F	=	1820	kN
Vacuum thrust IIInd stage	F	=	417	kN
Vacuum thrust III stage	F	=	133,4	kN
Weight of Ist stage at separation		=	1500	kg
Net weight IIInd stage		=	5500	kg
Useful-load casing		=	1000	kg
Insulation material of Centaur		=	570	kg
Net weight III stage with adapter, without WZ (expansion unavailable)		=	1910	kg

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3. INVESTIGATIONS CONCERNING THE PROPULSION SYSTEM OF THE SONDE

3.1 Highly energetic propulsion unit

3.1.1 State of Technology

As has already been discussed in Chapter 2.43, this propulsion unit is with regard to performance a very interesting solution and the best one.

Furthermore, preliminary projects for the development of cryogenic propulsion systems have been in progress in the Federal Republic since 1962, in Ottobrunn there is also a test stand for Fluorine, hydrogen engines, and experimental experience exists concerning the development of small fuel containers and cryogenic compound insulations. Furthermore, studies were conducted for ELDO over a period of two years concerning a highly energetic final stage.

Thus not only is the development of a (relatively small) propulsion unit well prepared, and is in accordance with the built-up capacity of experts and installations, but it also permits the practical employment of H_2/F_2 engines with 550 kp vacuum thrust which shall have been developed as of 1967.

In the USA there have hitherto been only studies concerning the design of kick stages, which are considerably larger and more

involved than is required here for a solar sonde or a Jupiter sonde.

3.12 Principles Design

A special characteristic of such propulsion unit is its extremely diversified employability, i.e. for different carrier rockets (ATLAS-CENTAUR, ELDO-B), as well as for different missions:

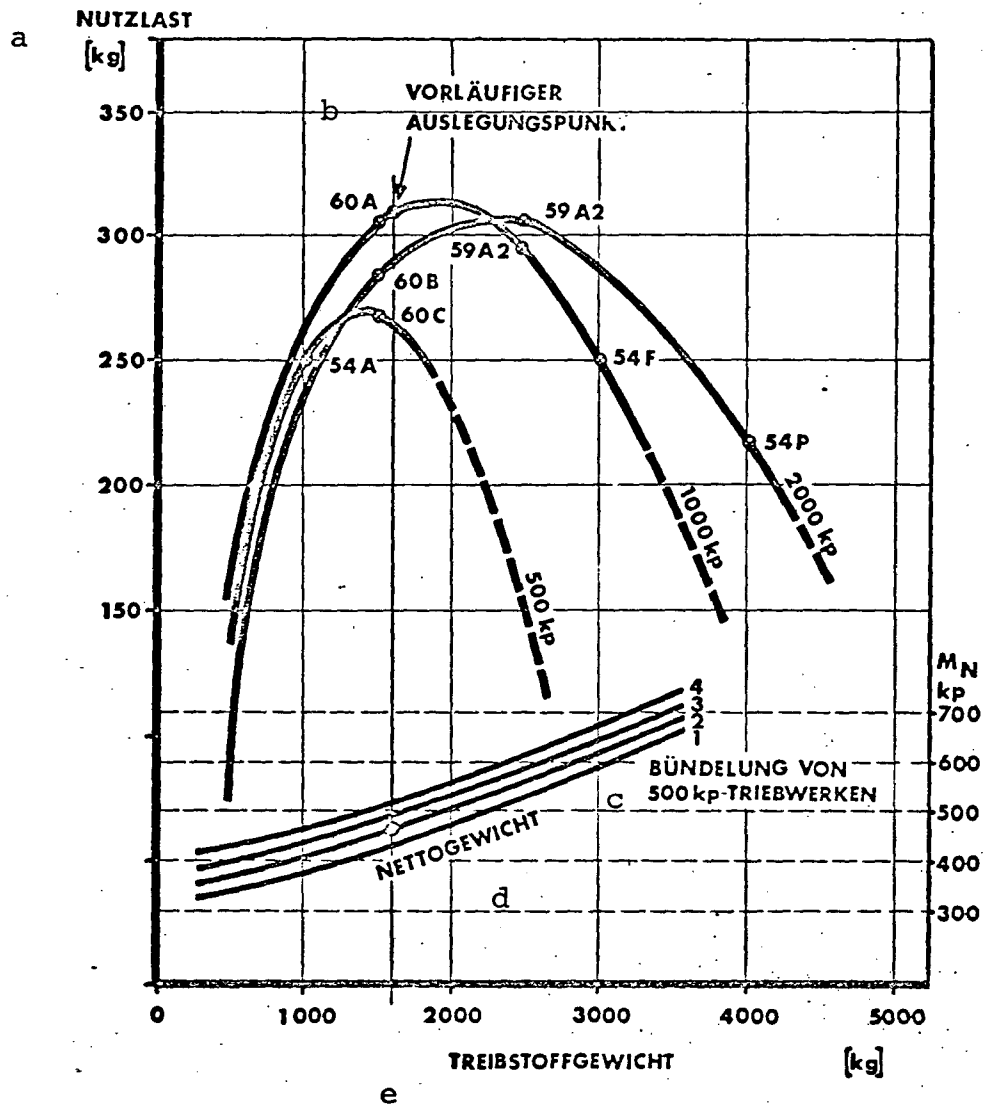
solar sonde
Jupiter sonde
comet sonde, etc.

The results of the optimization calculations necessary for determination of the thrust level and the fuel mass are shown in Fig. 3-01. Bundling of the planned H_2/F_2 engines, with 550 kp vacuum thrust for each, was here used as a basis.

The result for the present tasks (solar sonde with 0.3 and 0.26 AU perihelion and ATLAS-CENTAUR) is

1600 kg fuel, and
2 engines, each with
550 kp thrust.

FIG 3 - 01



Optimization of thrust and fuel
weight for HetAM

- a - useful load
- b - preliminary point of interpretation
- c - bundling of 500 kp engines
- d - net weight

With these data and those already obtained in previous project studies (OPHOS - I - series), it is possible to work out a survey of configurations (FIG. 3-02).

A comparison of the configurations, which is described in detail in TN-P6B-23/66, leads to the result that Design D represents the optimal solution, namely optimal with regard to weight, reliability and development cost.

Moreover, this type of construction can be easiest adapted to different fuel weights.

A construction design was prepared for this version with four parallel tanks (FIG. 3-03), a weight estimate was also made.

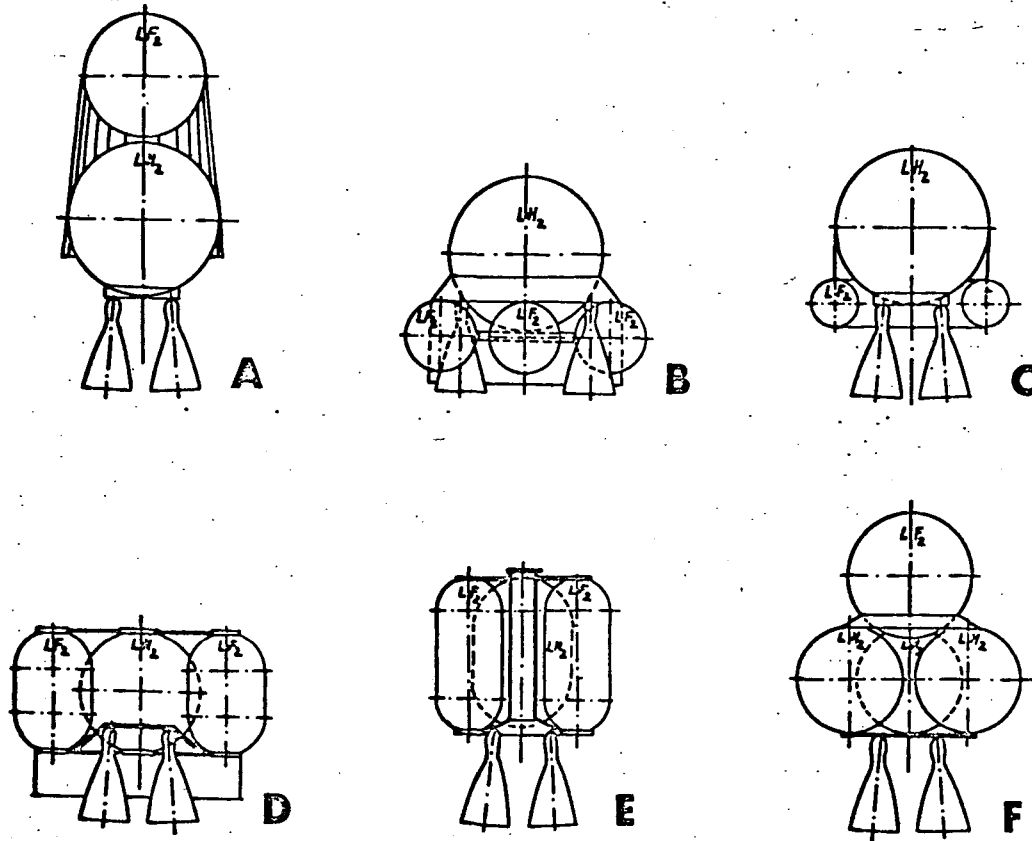
The results are shown in TABLE 3-I.

TABLE 3-I: Weight Estimate for a Propulsion Unit with 1.6 y

Fuel (H_2/F_2)

M₂ ELECTRONIC EQUIPMENT	
Gyroscopic-platform vehicle-born computer	20,0 kg
Velocity gyroscope and electronic guidance equip.	3,5 "
Command device	5,0 "
Measuring-value giver & amplifier	15,5 "
Decoder and data processing	10,0 "
M₃ ENGINES AND SUPPLY SYSTEM	
Main engines (2)	39,8 kg
Helium system	38,8 "
Fuel fittings	11,0 "
M₄ TANKS AND CELL	
Fluorine tanks	35,1 kg
Hydrogen tanks	58,5 "
Insulation	7,9 "
Other structure	26,0 "
M₅ POSITION REGULATION/ELECTRONIC EQUIPMENT	
Regulation electronics	3,5 kg
Secondary engine system	18,2 "
Preacceleration system	3,0 "
Pneumatic system	5,5 "
Electric equipment with batteries	33,0 "
M₆ AUXILIARY EQUIPMENT	8,0 kg
M₇ AMOUNTS OF REMAINING FUEL	
Fuels for position regulation, evaporated fuels (portion of 30%)	72,1 kg
M₈ OTHER WEIGHTS	
including 10% safety margin	47,6 kg
	<hr/> 462,0 kg.

FIG 3 - 02



Investigated tank configurations
for HetAM with 1600 kg fuel (H_2/F_2)

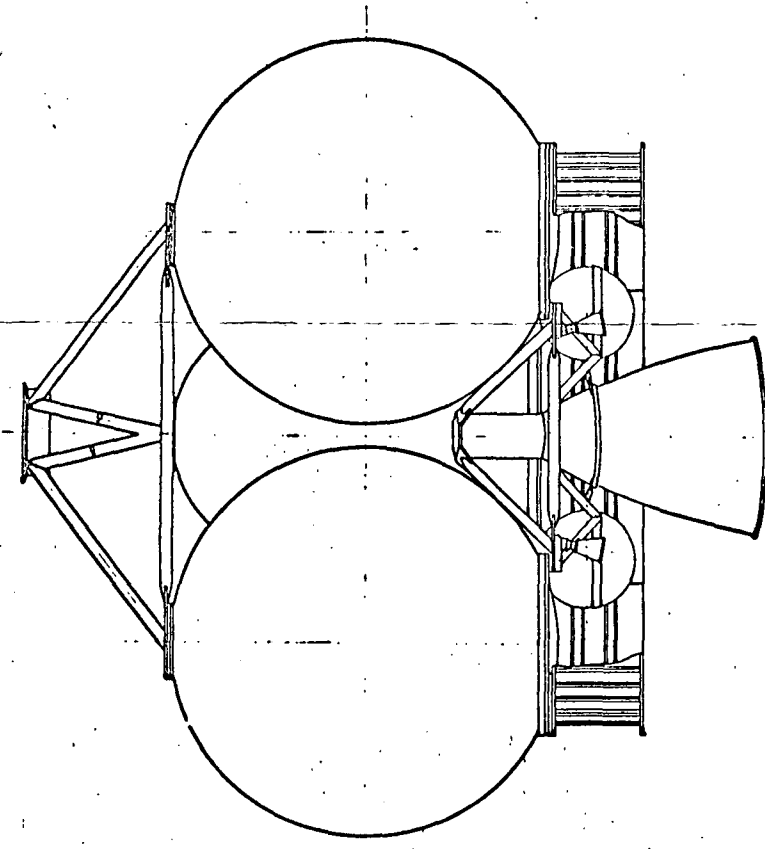
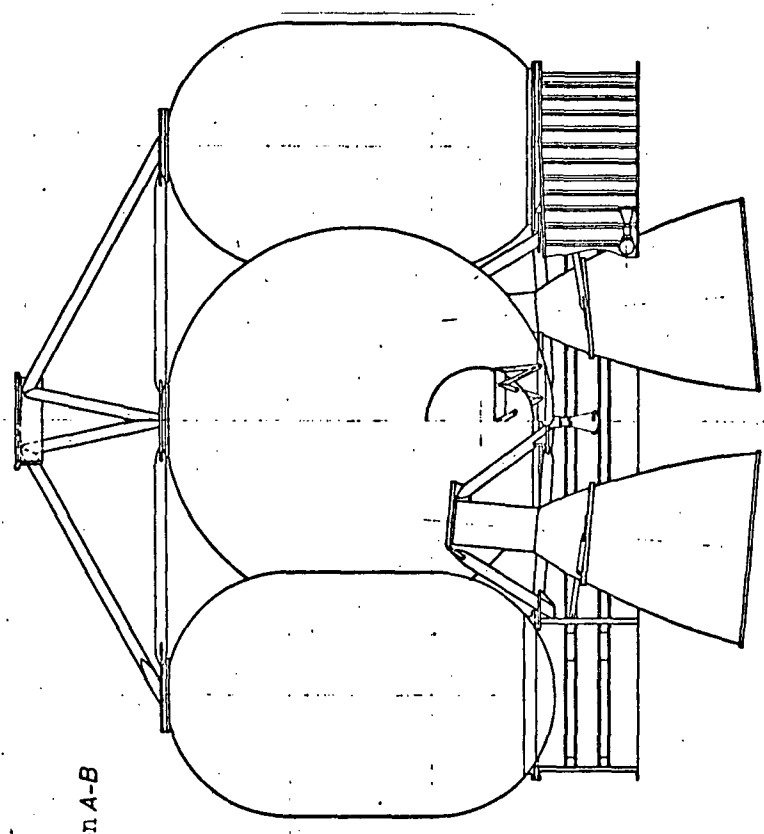
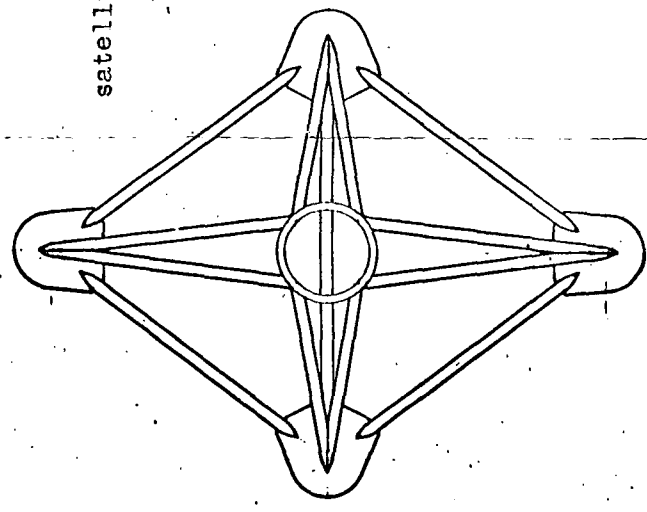
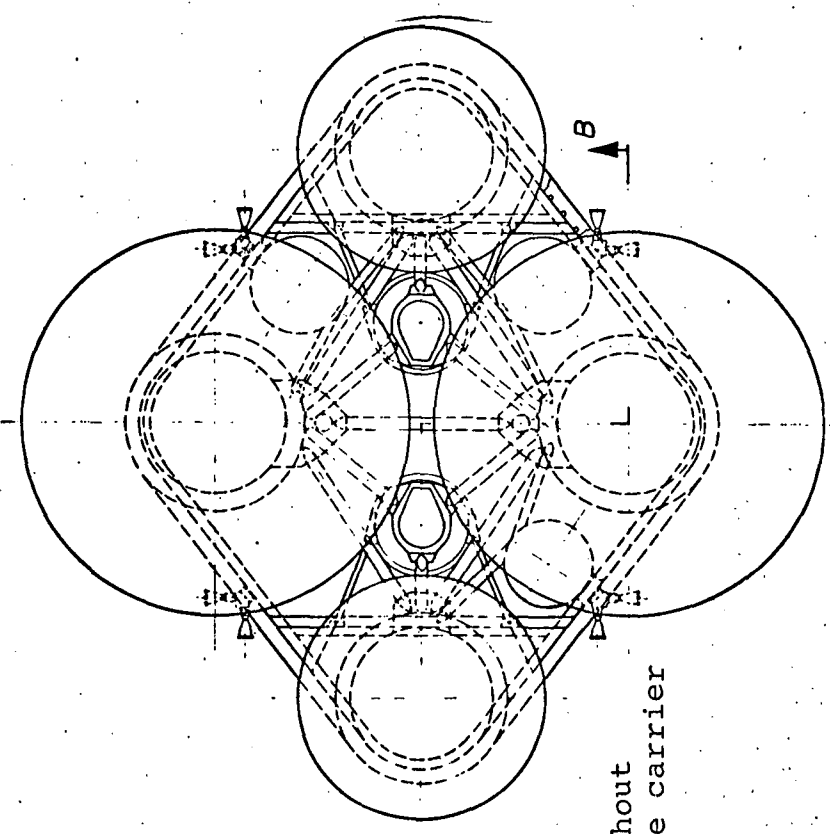


FIG 3-03



Construction
Design of the
highly energetic
propulsion unit
(HetAM)

The device consists of two hydrogen tanks each 1.33 m in diameter, and two fluorine tanks, each having a diameter of 0.85 m. The height of all four tanks is 1.35 m. They are connected by an upper and lower framework, which simultaneously serves for mounting of the useful load and of the two engines.

The advantages of this solution are:

- High safety and reliability due to separate fuel containers;
- Convenient accessibility and accommodation possibilities for equipment and installations, easy mounting and dismantling possibilities;
- Low construction height and good volume flexibility with small net weight.

Easily weldable aluminum alloys shall be used as material (AlZnMg 1); the "compound insulation" of hard polyurethane foam with Super Insulation, at present being developed at the Boelkow firm, is suitable for the tank insulation. The undesirable fluorine evaporation prior to launching or during climbing can also be impeded by this insulation (at least for a prolonged, normally sufficient period of time).

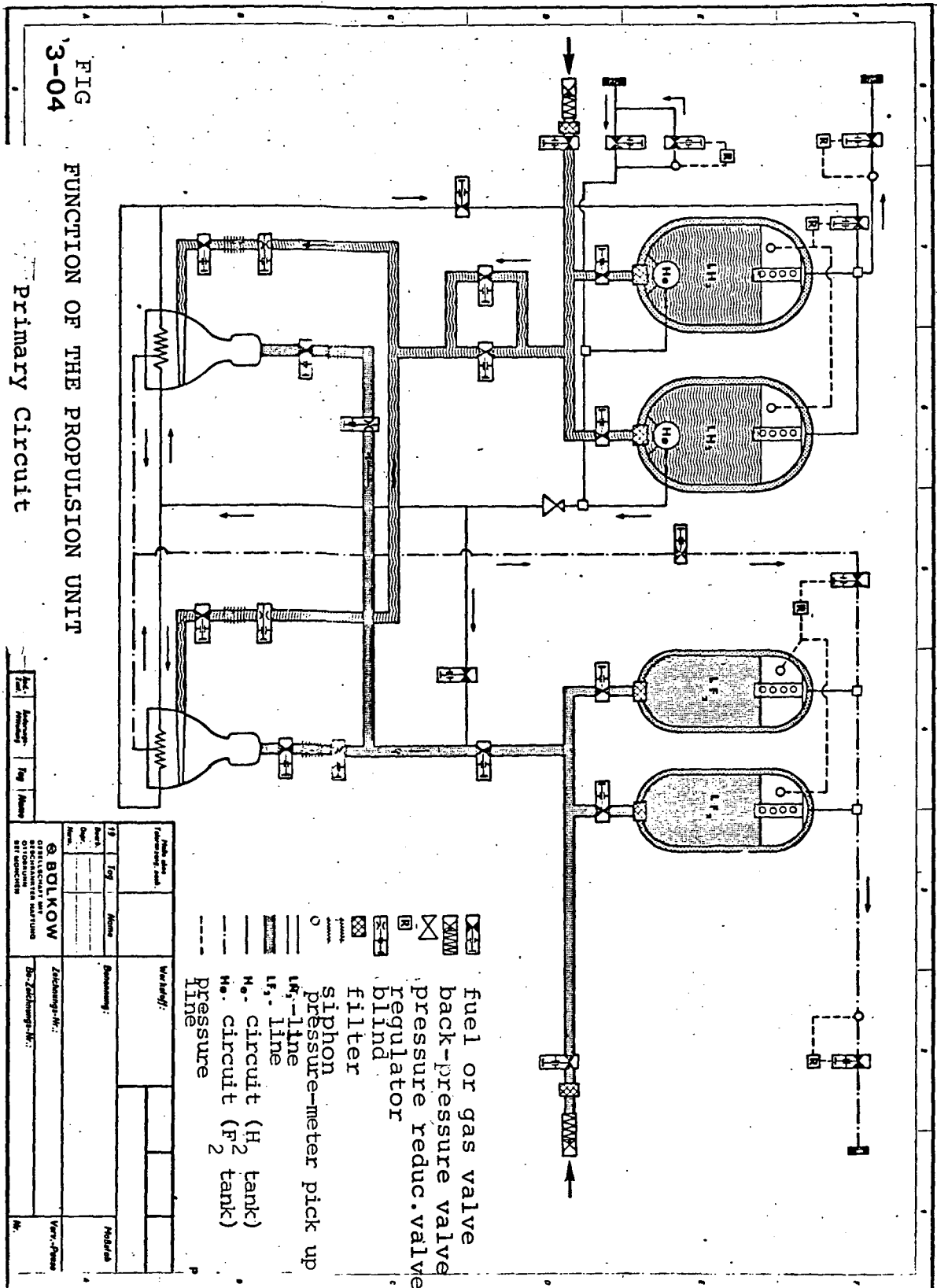
The delivery of the fuels can and shall be effected by means of pressure gas. A detailed analysis of the possible systems (see TN-P6B-48.66) led to the choice of helium for both tanks, which is stored in the hydrogen and heated to 300° K and 120° Kelvin.

Two H_2/F_2 engines with 550 kp vacuum thrust (or 300 kp ground thrust), to be developed in 1967, shall be used as the propulsion system. A detailed design study, upon which the design data of the propulsion unit are based, has already been worked out at the Boelkow firm.

The problems of position regulation of the HetAM device were also investigated, since it is planned to install the two main engines rigidly. In this case an additional system with small engines must be provided which makes position regulation possible about all three axes. Here thrust values of 2.2 to 3.5 kp were yielded, based upon the following moments of perturbation:

axis of rotation	2220 kp sec
axis of yaw	4300 kp sec
axis of pitch	4950 kp sec

This secondary system could, for reasons of reliability and reduced development cost, employ the 3-kp engines with "aerozin" $50/N_2 / O$ which were already tested in Lampoldshausen.



For reasons of simplification and weight economy, the conceived HetAM device does without a separate transponder and telemetry sonde, since the corresponding devices of the sonde can be used at the same time.

3.2 Radioisotope propulsion unit (RAMSES)

Such a propulsion system consists in the heating of hydrogen by the decomposition heat of radioisotopes. Since this effect can also be made use of for the generation of electric energy, via thermoelectric or thermoionic converters, it is obvious to combine these possibilities (similarly to the SNAPOODLE concept of the American firm TRW Systems).

Such a propulsion system is especially interesting because the thermal energy is made use of to a great extent. In the case of a radio nuclide generator, only approximately 5% of the heat is converted into electric energy. The attainable specific impulse is limited only by the temperature stability of the materials used. The upper limit is today at 800 sec. specific impulse and 2000°K hydrogen temperature. Since in outer space a high expansion ratio can be easily attained, the pressure in the engine is of subordinate importance.

The theoretical relationships between pressure, temperature, and

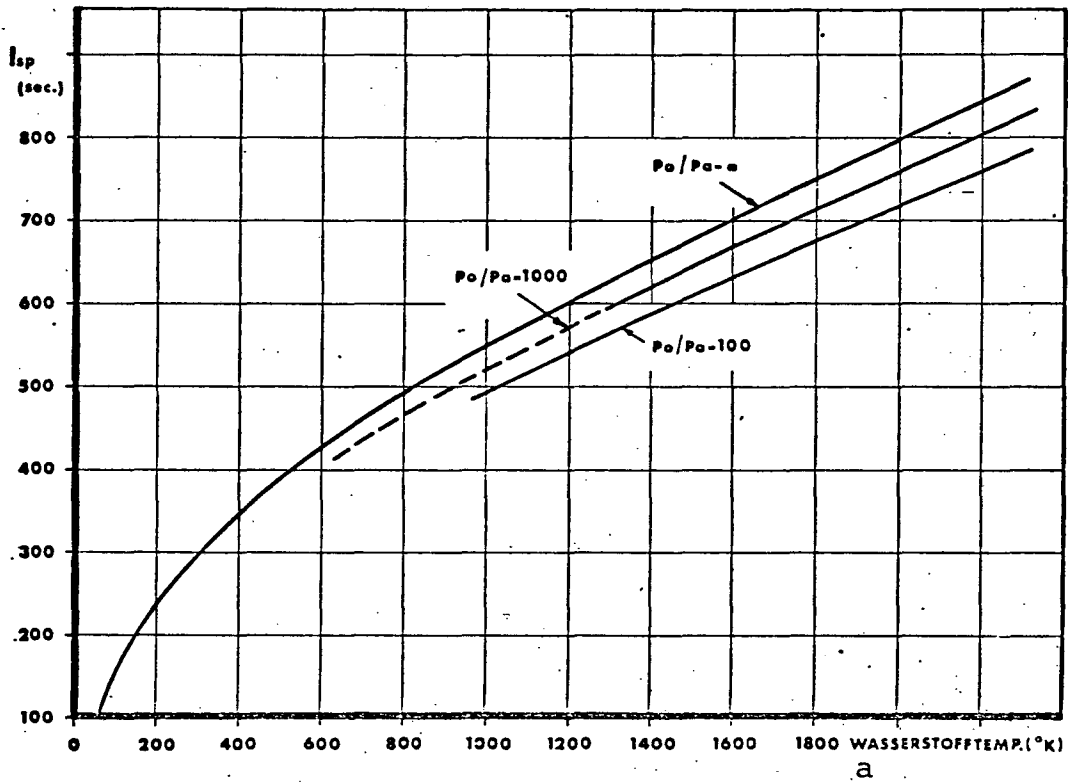
specific impulse for hydrogen as propulsion gas are represented in FIG. 3-05.

A propulsion system of this kind is very simple from the point of view of functional technology. A delivery system is not necessary. Only a safety system for the tank is required. The heat entering from outside causes the liquid hydrogen to evaporate, whereby a pressure is built up. By means of expedient design of the insulation, the result is attained that the pressure in the tank changes only little during the propulsion phase. With this simple method an almost constant thrust consumption and specific impulse are obtained.

During and prior to launching the generator already generates heat, which has to be conducted off by means of a cooling circuit. Thus oxidation of the channels and nozzle is avoided, and the unit is kept at a low temperature. If the launching should fail, the encapsulated radioisotope must be protected against glowing upon re-entry into the atmosphere. Moreover the casing may not burst open upon hitting rock, nor, upon striking the ocean, become leaky during several half-life periods.

These safety problems in connection with the high working temperature planned indicate the difficulties connected with the develop-

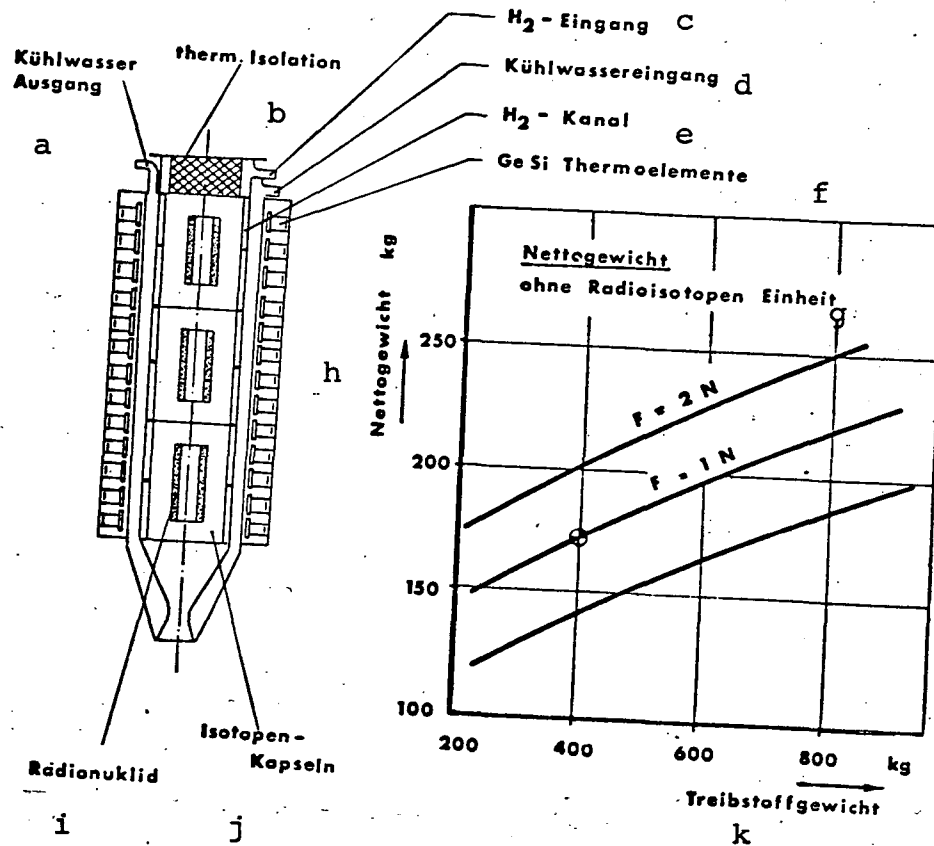
FIG 3 - 05



Specific impulse of heated hydrogen as
function of temperature and expansion
ratio.

a = hydrogen temperature

FIG 3 -06



Net weight of RAMSES as function of the fuel mass, and schematic drawing of the radioisotope generator with a H₂ nozzle.

a = cooling-water discharge, b = thermal insulation
 c = H₂ entry, d = cooling-water entry, e = H₂ channel
 f = GeSi thermoelements, g = net weight
 without radio isotope unit
 h = net weight kg, i = radio nuclide, j = isotope
 capsules, k = fuel weight

ment of radionuclide drive.

Since hydrogen possesses a low specific gravity, the fuel tank already becomes rather large and heavy already at low weights.

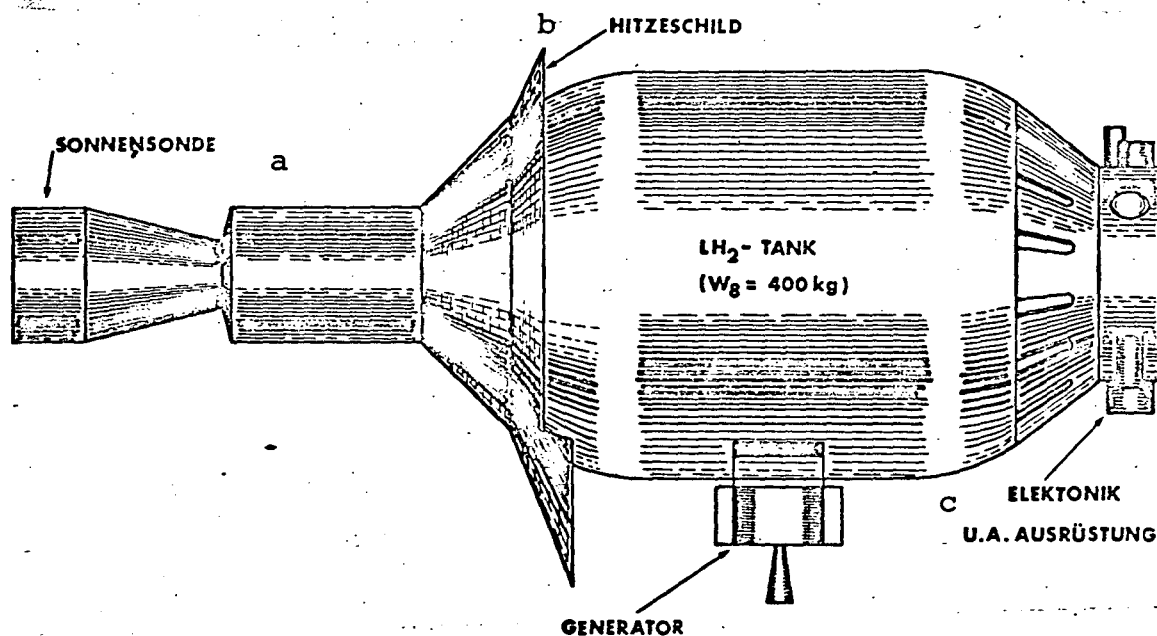
In addition to a functional diagram, FIG 3-06 contains the net weight as function of the fuel weight. A first design was made for 400 kg hydrogen. This value resulted from the optimization of the orbit mechanism (compare FIG 2-18). Since the thrust shall exert an azimuthal effect, but since the sonde must be always oriented toward the sun, it is necessary to mount the engine perpendicularly to the longitudinal axis. At the same time, care must be taken that the thrust vector always point through the center of gravity in order to avoid moments. In addition, it must also be pointed out that the waste heat of the propulsion unit must be radiated, i.e. the generator must have a free field of vision. In order to comply with these limitations, a cylindrical tank, sealed with hemispherical bottoms, was placed at the center of gravity. The electronic equipment, as well as the remaining equipment of the unit, was arranged on the longitudinal axis, away from the sun. In order to protect the tank from direct solar radiation, a heat shield was provided, in the shadow of which the engine is also situated. The corresponding configuration is represented in FIG. 3-07. An engine of this type was tested

in February 1965 during a three-day test period in the AEC Mound Laboratory (Ohio), and yielded specific impulses of 650 to 700 sec (thrust approximately $1N \approx 0.1$ kp). With more progressive versions, it is expected that later on performances of up to 1200 sec I_{sp} can be reached.

The choice of the radio isotope is influenced by the mission, a compromise must be made between specific gravity, half-life period, and the necessary shielding weight.

Curium Cm242 was chosen as heat source for the RAMSES drive. Of all the available radioisotopes with high power density and, at the same time, low shielding requirements, the costs are lowest in the case of Cm 242 (approximately 1 million dollar for 6 kW_{th}). Moreover, the half-life period of this isotope (163 days) is very well adapted to the duration of the mission. For, during the entire drive phase (30 - 50 days) the performance decreases only little, whereas approximately 40% of the isotope capacity is conducted to the thermoelectric converter system. When the perihelion distance is reached approximately 60% of the initial capacity of the heat source still remains available.

This is now conducted solely to the thermoelectric converter system, as has already been done during the entire free-flight phase, so that a 50% higher capacity than in the drive phase is obtained.



Design diagram of a solar sonde with
radioisotope propulsion unit (RAMSES)

a = solar sonde

b = heat shield

c = electronic & other equipment

3.3 Nuclear-Electric Propulsion Unit

The third possibility for a sonde propulsion unit is the employment of electric engines.

Although the state of development of these systems hardly permits immediate employment in a project to be realized in the next few years, this possibility shall be investigated, in order to find out the extent to which it should be pursued for subsequent solar sondes or similar projects.

The results of the performance calculations were already briefly mentioned in Chapter 1.4; they are discussed in detail in TN-P6B 46/66.

Basically, the orbit mechanism of electric drives is very complicated, so that the projects carried out in this field over a period of two years by the Boelkow firm commissioned by the GfW were of great use.

However, the results are quite disappointing; only under special conditions was a performance advantage yielded in comparison to the chemical drives, and only for considerably longer flight periods. This is due to the fact that the energy supply becomes relatively

difficult even with the latest project data of the Interatom/Siemens Firma (Fig. 3-08).

It can be seen from the diagram that, in practice, only reactors with fully thermionic energy conversion can be considered ($\alpha = 38 \text{ kg/kW}$).

In the case of capacities up to 45 kw, which are sufficient for the solar sonde, even energy generation via solar cells is more advantageous with regard to weight (and more reliable;) than a reactor system. The firm Boeing is at the present time developing a solar cell system (commissioned by NASA) with 116 m² area and 12.5 kw capacity. It will, however, be difficult to make these surfaces adjustable as is required for the solar sonde.

In each case, however, long flight periods are yielded, as shown in FIG 3-09. Here are represented the results of variation calculations with parameters:

- specific impulse 300 to 8000 sec
- electric capacity 40 to 80 kw;

they show flight periods between 200 and 1000 days (in comparison to 90 days in the case of chemical drive). These values are with

reference to an initial orbit altitude of 500 km (with 4000 kg mass). In addition, the initial orbit altitude of 300 km was investigated. In this case, it is true, higher useful loads are yielded, but also longer flight periods.

In addition to the critical capacity results and the questionable reliability during the long flight period, there exists also a series of technical difficulties:

- (a) The reactor system alone requires 70 to 80% of the initial mass of 4000 kg; the shielding requires (according to the present concept) a distance from the electronic systems of at least 15 m. This can lead to construction lengths of more than 60 m, which cause considerable launching difficulties with regard to the carrier system. Shorter construction lengths call for a stronger shielding, and thus more weight.

Furthermore, in each case a kind of telescopic tube with an extension mechanism must be provided (compare FIG. 3-10).

- (b) The thrust vector of the ion engine must be adjustable within a large range, which is practically possible only by means of a revolvable arrangement of the engine

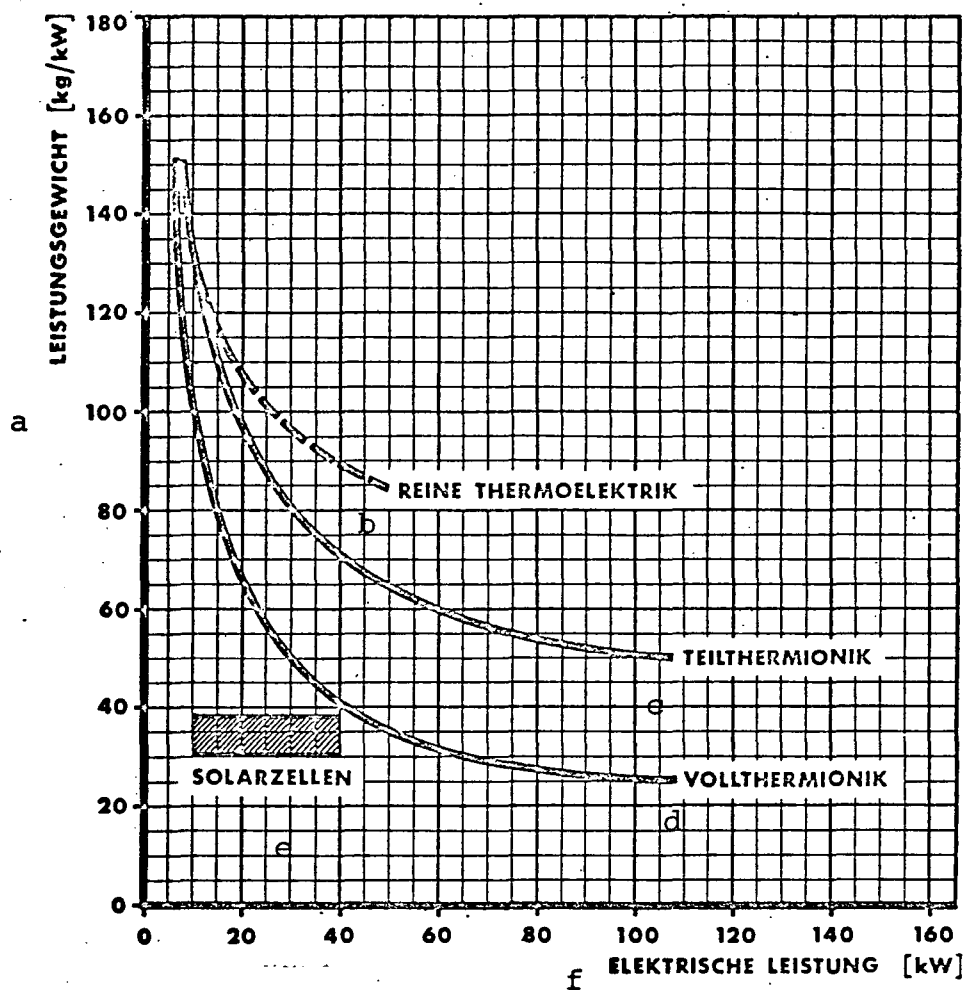
palette. The fuel tank must also be mounted on this revolving body, which must be exactly in the center of gravity of the system, in order to maintain the center of gravity of the system in spite of fuel consumption.

This difficulty could be avoided only if radial thrust were to be assumed over the entire orbit (FIGURE 3-10); this, however, would involve considerable loss of efficiency.

Mercury or cesium are considered as fuels for the electric engines; the required amount is approximately 1 ton. If all the ion engines developed today are used as basis, approximately 30 engines (specific weight 0.5 kg/kw_{el}) are required for a solar sonde for 60 kw capacity and 6000 sec specific impulse.

It can thus be concluded that nuclear-electric drives are practically out of the question for the "solar sonde" project; it also remains open, whether this concept is meaningful at all for interplanetary sondes, and whether it is worth while for this purpose to develop a reactor which requires an outlay of more than 500 million DM, and which is presumably insufficient with regard to its specific weight.

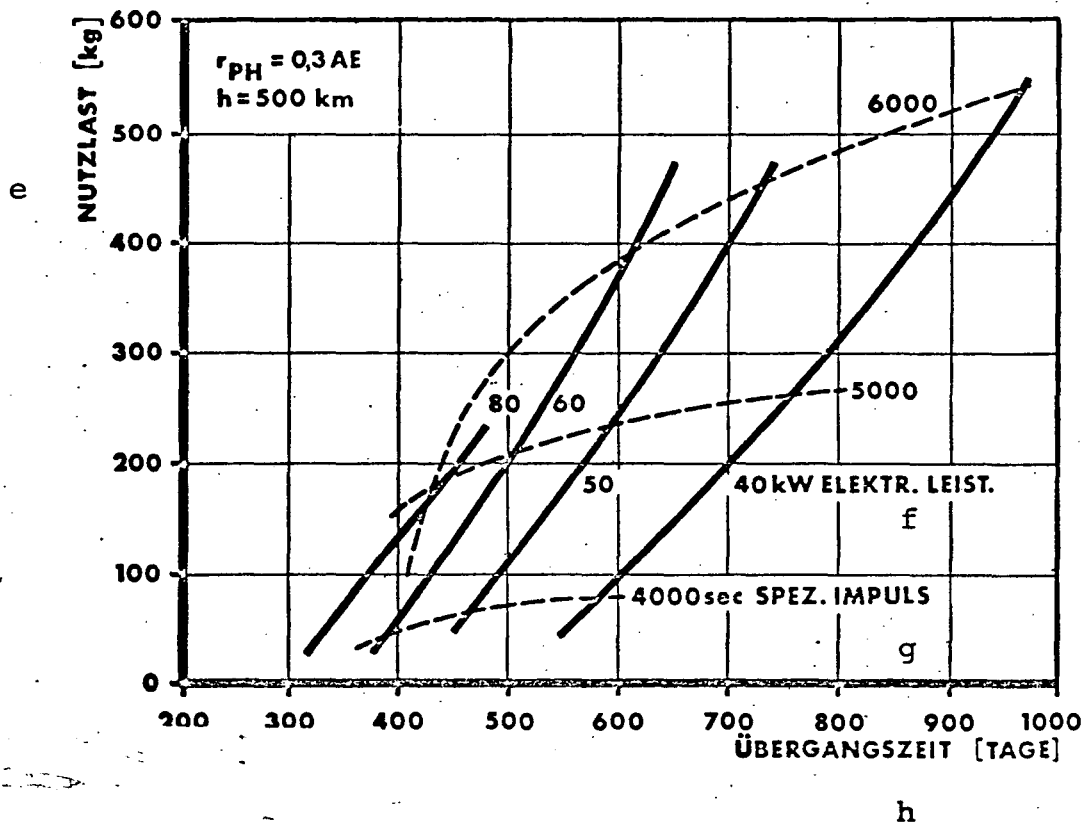
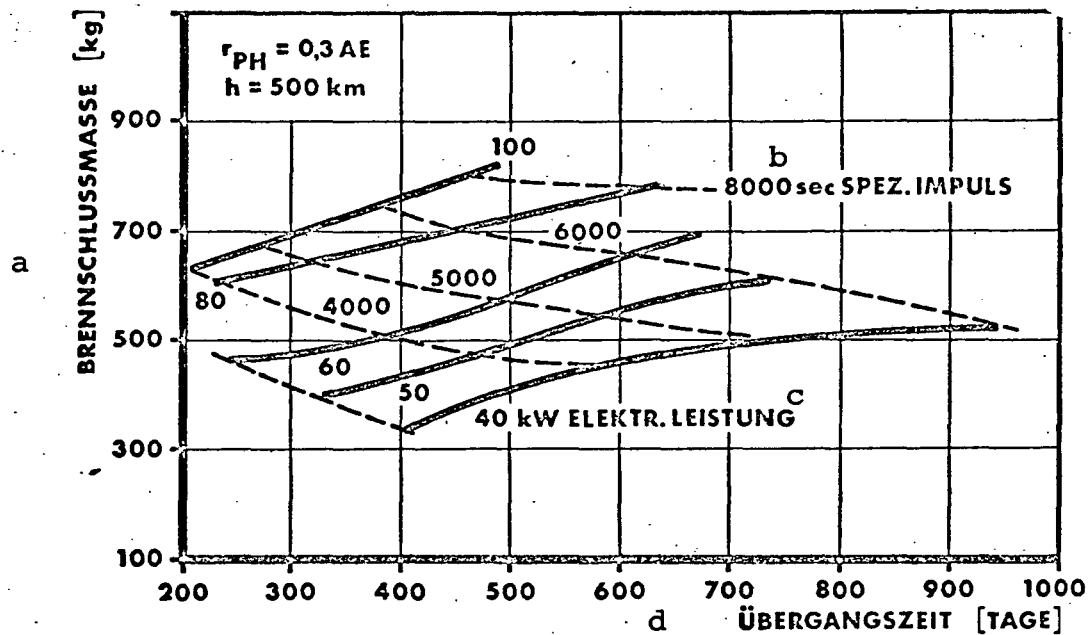
FIG 3-08



Specific weights of reactors with thermoelectric, partially thermionic, and fully thermionic energy conversion (according to SW/INTERATOM):

a= specific weight b= pure thermoelectric
 c= partial thermoelect. d= full thermoelectric
 e= solar cells f= electric capacity

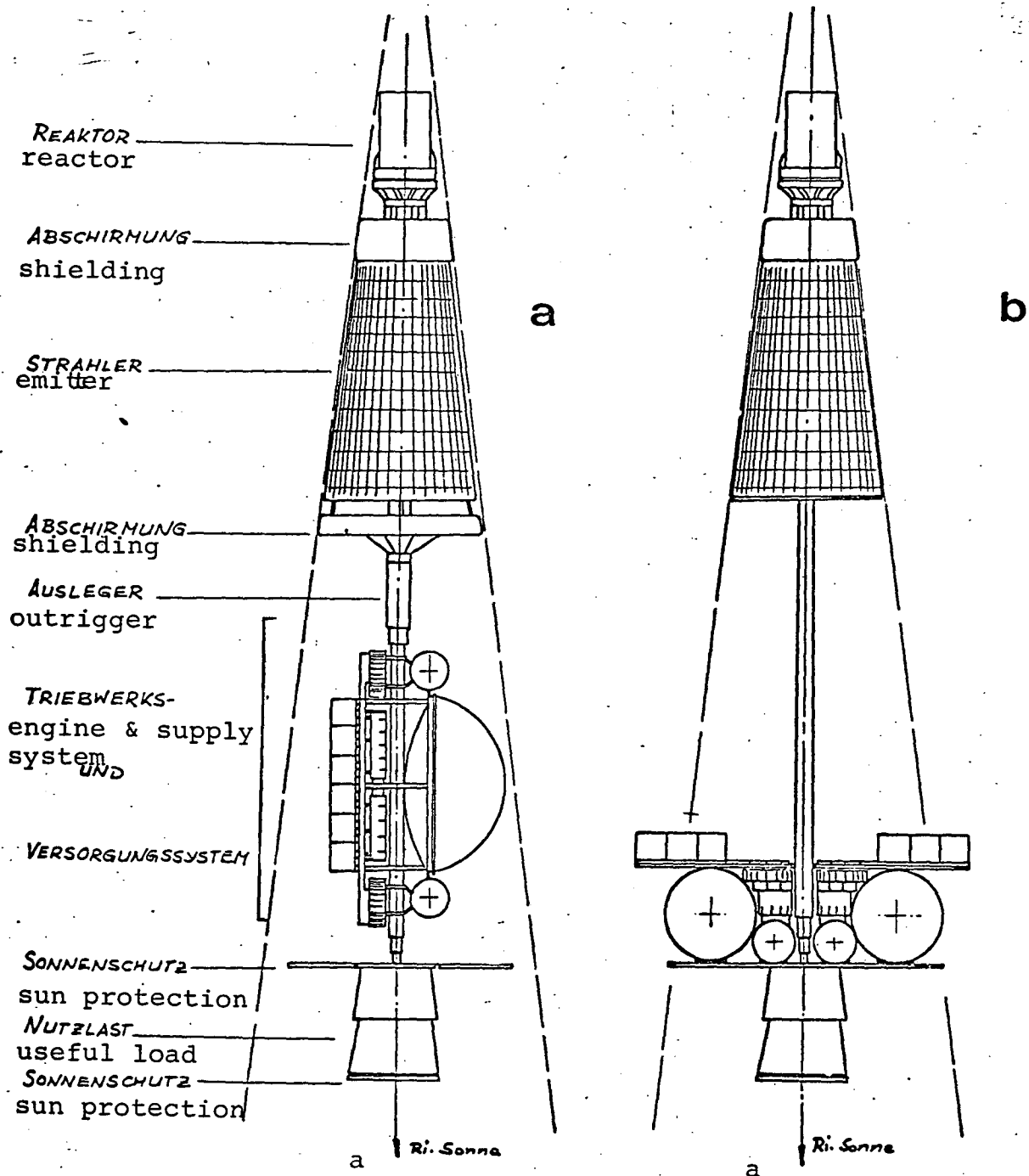
FIG 3-09



Results of capacity analysis of electric drives for a solar sonde: useful loads and final masses with specific impulse and electric capacity requirement as parameters.

a= combustion cutoff mass, b= specific impulse, c= elec. capac.
d= transition period (days), e= useful load, f= elec. capac.
g= specific impulse, h= transition period (days).

FIG 3 - 10



a= direction to sun

4. OPTIMAL POSSIBILITIES FOR THE LAUNCHING OF A SOLAR SONDE

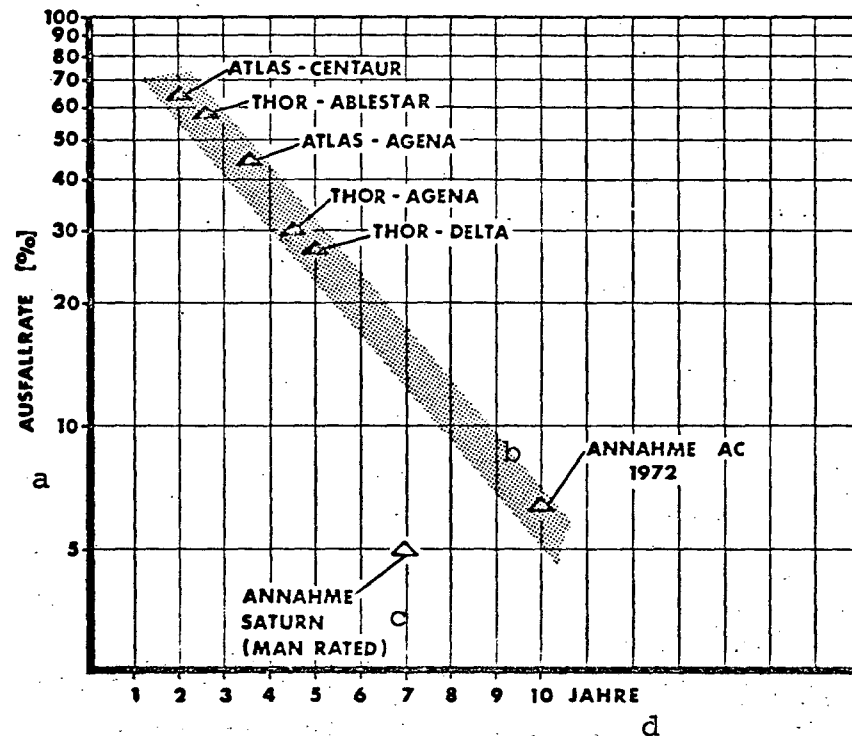
4.1 Reliability and Cost

The efficiency analysis, chapters 2,4, has shown that there is a series of possibilities of bringing a solar sonde with approximately 150 kg mass into orbit with < 0.3 AU perihelion. For determination of the optimal possibility (ies), cost and reliability must, however, be taken into consideration.

The reliability of today's carrier rockets and carrier stages is known, and can also be extrapolated. In FIG. 4-01, a statistical model for the course of reliability (decrease of the failure rate) is represented as function of time.

Investigations have shown (compare TN-P6B-35/66) that the failure rate depends not so much upon the number of launched devices as rather upon the period of operation. For recent developments, especially for carrier systems which were developed with regard to highest safety requirements for manned space flight (as SATURN I B and SATURN V), an overall reliability of 95% may be assumed. In the case of the development of the BURNER - II stage, for instance, a reliability proof of 96% was required from the very beginning, even before the device was admitted to flight tests.

FIG 4-01



Increase of reliability as function of the period of employment for carrier rockets

a = rate of breakdown

b = assumption AC 1972

c = assumption Saturn (man rated)

d = years

In order to determine the number of carrier rockets required in the various cases (and thus the cost of launching), a basic determination of the required or desired mission reliability must be made; i.e. with what degree of probability is the "solar sonde" mission to succeed.

It is here assumed that the mission shall be carried out with

86% probability of success.

This goal can, for example, be attained with one SATURN-V carrier rocket, which launches two sondes simultaneously, the individual reliability of each of which is 80% .

Computing with the same reliability data, the result is then obtained by means of comparison that two launchings of the SATURN IB-CENTAUR are necessary, and even four launchings in the case of the SATURN IB-ELDO-III-BURNER II combination.

The employment of ATLAS-CENTAUR would, on the other hand, require two to three launchings, in accordance with the reliability which may be assumed for the additional propulsion unit. In the case of two launchings it should have approximately 85% dependability, for three launchings only 65%. This shows clearly that approxi-

mately 16 million dollars of launching costs can be saved if the reliability of the propulsion unit is raised from 65 to 85%, which will presumably not require an outlay of more money, but will require a longer period of development. The launching costs of the different carrier rockets are approximately as follows:

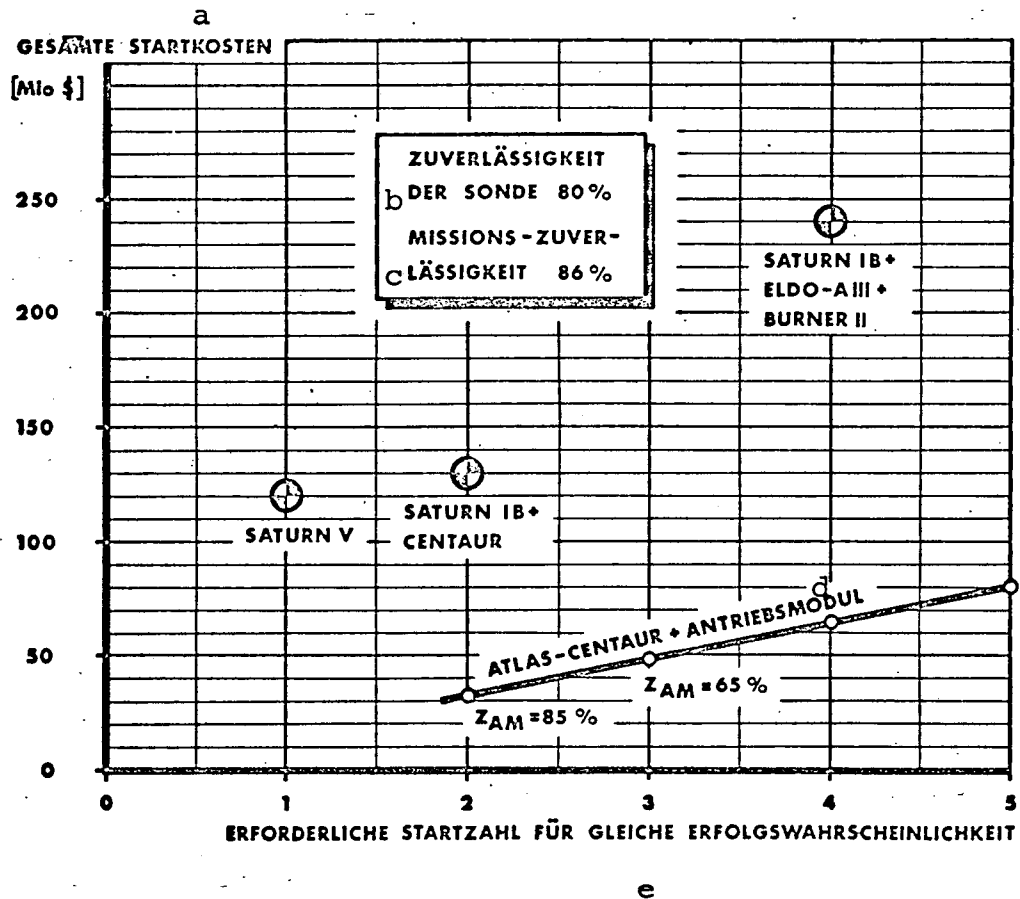
Atlas-Centaur	14 million dollars
Saturn - IB	55 million dollars
Saturn - IB -Centaur	aprox. 65 million dollars
Saturn V	125 million dollars.

If supplementary stages are used, these costs will rise somewhat.

The result of these considerations is represented in Fig. 4-02: the total launching costs in the case of employment of different carrier rockets. The figure shows the following basic results:

- (1) The employment of SATURN IB with supplementary stages can be hardly justified from the point of view of cost.
- (2) The employment of the SATURN V carrier rocket requires only a single launching (with two sondes simultaneously), and is also the safest and quickest solution.

FIG 4-02



Comparison of launching costs for 4 carrier systems under the assumption of identical probability of success of mission

- a = total launching costs
- b = reliability of sonde 80%
- c = mission reliability 86%
- d = propulsion unit
- e = required number of launchings for the same probability of success.

- (3) The least expensive solution (from the point of view of launching costs) is the employment of ATLAS-CENTAUR; it requires, however, the development of an additional propulsion system, upon the reliability of which again depend the number of launchings and the total cost.

The launching costs are, it is true, least in the case of the employment of ATLAS-CENTAUR, but it must not be overlooked that there are two additional cost factors:

- (1) cost of development of the propulsion unit,
and
- (2) cost of the prolongation of the entire program
in the case of two or three launchings (program
prolongation by three to six months -- or
more).

The total costs can therefore certainly become comparable to the cost of a SATURN V launching.

4.2 Selection of a Propulsion Unit in the Case of Employment of ATLAS-CENTAUR

If a decision is made in favor of ATLAS-CENTAUR, it must also be decided what additional propulsion unit should be used. In connection with this the following statement is made:

- (1) It is unlikely that an electric propulsion unit (also with solar cells for energy generation) will be ready for operation in less than 8 years, especially if the reliability requirements for the long flight period are taken into account. Furthermore, it is likely that the cost of development will amount to several hundred million DM; it can at present not yet be estimated, not any more than the reliability of such a system. For these reasons it is practically impossible to consider such a system for the solar sonde project (for which advantages of performance probably cannot be expected).
- (2) A radioisotope drive is, true enough, connected with a somewhat smaller development risk than is the solar-electric drive, the employment of the nuclide generator as such is, however, afflicted with certain safety problems, which hitherto have not been completely clarified.

Since it was shown that no nuclide generator is necessary for energy supply in the case of the solar sonde project (up to 0.26 AU), it is probably also not expedient to provide this system in this case. Moreover, the performance is not particularly high and has not been verified in space for prolonged flight periods.

- (3) The third possibility of a highly energetic propulsion unit is the solution which can be most safely mastered with regard to technology. The development cost and reliability values can be determined for it; moreover, the capacity or useful load is the highest of all comparable systems.

4. 3 ATLAS - CENTAUR - HetAM

This solution is connected with the lowest launching costs, but requires the development of a highly energetic propulsion unit (HetAM).

This propulsion unit is no complete stage (or need not be one); it can, for example, also remain connected with the sonde. In each case it employs systems of the actual sonde, so that the weight of the unit is smaller than the weight of a complete stage.

The cost of development of such a unit with 1.6 t fuel weight amounts to approximately 195 million DM (without engine development), and is relatively small due to the preliminary work already performed.

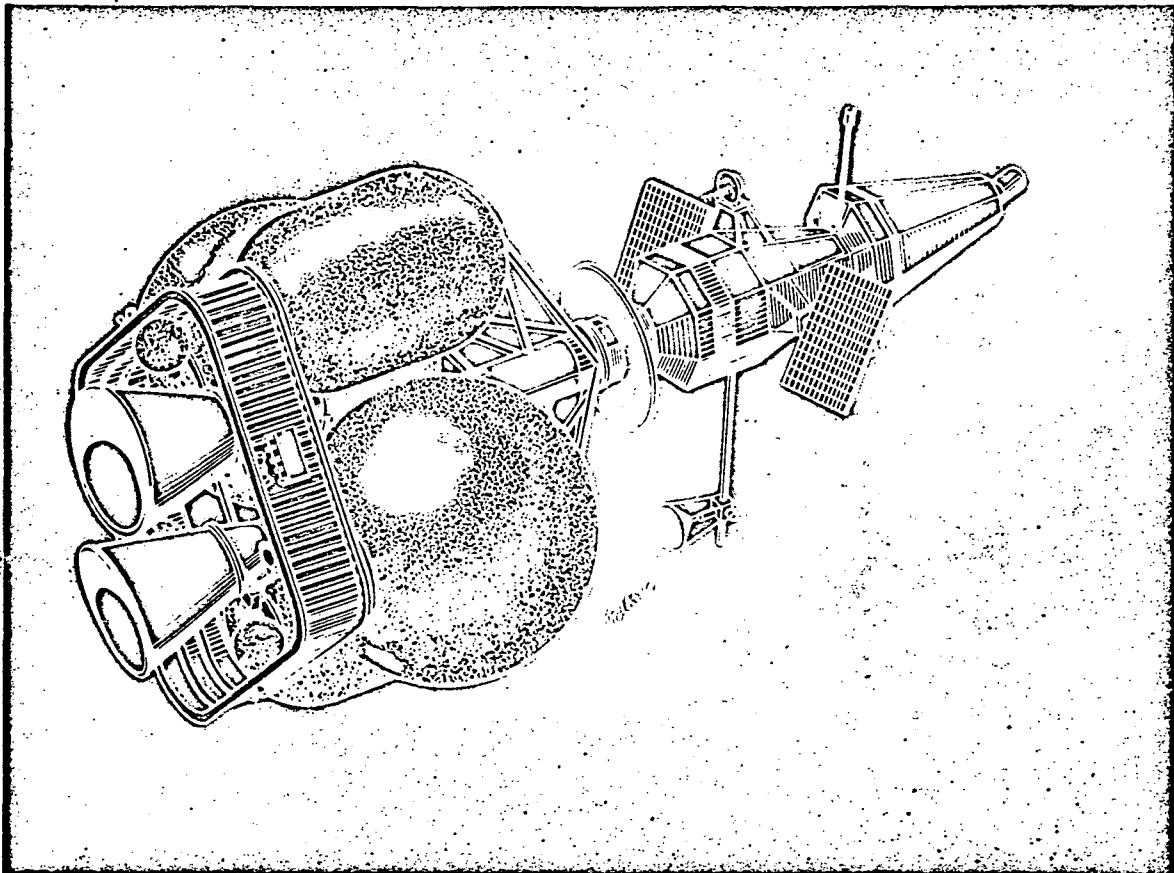
This also has an effect upon the development period, which was determined to be 4 to 5 years. HetAM can, incidentally, also be employed with small changes for the launching of other space sondes, not only for the solar sondes.

The design concept of the HetAM has already been described in Chapter 3.1.

FIGURE 4-03 shows a perspective representation of HetAM solar sonde, FIGURE 4-04 shows a detailed drawing with explanation. At least two launchings must take place, in order to have a high probability of success. In the case of launchings in quick succession there is, moreover -- in the case of success of both starts -- the possibility of obtaining experimental data from two points of the orbit, which considerably increases the value of the results.

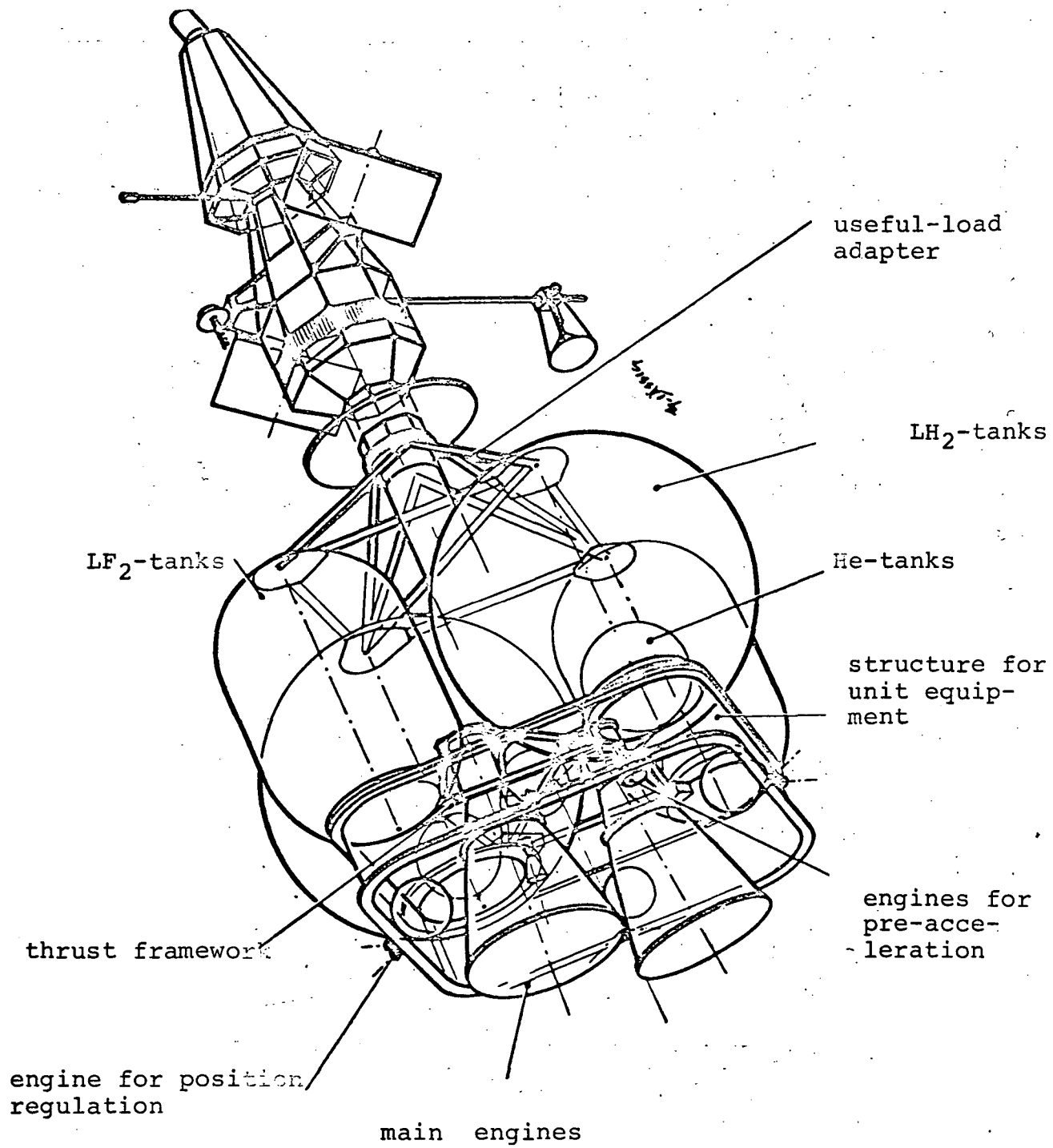
FIG 4-03

Reproduced from
best available copy.



Solar sonde with highly
energetic propulsion unit
(HetAM)

FIG 4-04



4.4 SATURN V with Double Launching

The most simple, most reliable and quickest solution for the launching of a solar sonde is doubtless the employment of the SATURN V carrier rocket.

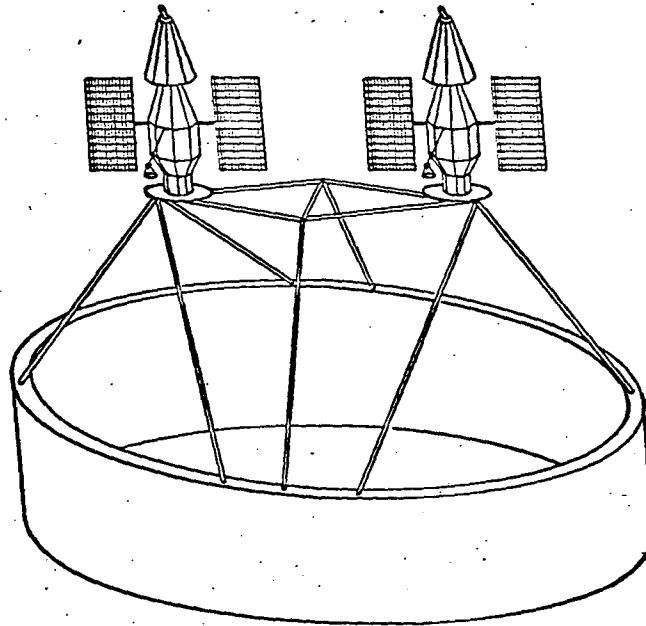
In this case, two sondes can be launched simultaneously; these according to the performance diagram (FIG 2-15), may possess an almost arbitrarily large mass, i.e. may have an extended measurement program and can come as near the sun as 0.23 AU.

The reliability and the probability of success of this concept may be considered optimal in every respect, since"

- (1) SATURN V is "man-rated", i.e. is designed for highest reliability due to the manned moon mission.
- (2) The reliability is not diminished by any additional stage or by a supplementary propulsion system.
- (3) In the case of the simultaneous launching of two sondes, complete redundancy is present.

Moreover, the value of the experiments is thus (by simultaneous measurement of two points in space) considerably increased.

FIG 4-05



Parallel launching of two solar
sondes with SATURN V

5 ORBITS EARTH - SUN

5.1 Injection Conditions and Free-Flight Orbits

The injection conditions and the required thrust for solar orbits with a perihelion from 1.0 to 0.2 AU have already been discussed in Chapter 2.2.

Report TN-P6B-40 66 contains supplementary information. Special injection conditions are furthermore demonstrated by a concrete example in Chapter Five, 3.

It remains here to discuss the free-flight orbits which are yielded in accordance with the injection conditions, or those which the conditions require.

The orbit region of interest, with a perihelion distance of 0.3 to 0.25, is represented in FIGURE 5-01. The flight periods (in days) are also noted. We are dealing here with the conventional representation of an undisturbed elliptical orbit with a fixed coordinate system.

Another representation may, however, be chosen, in which the coordinate system rotates with earth velocity. In this case a better survey is gained over the periods in which the sun stands in the way of a radio sonde-earth radio connection,

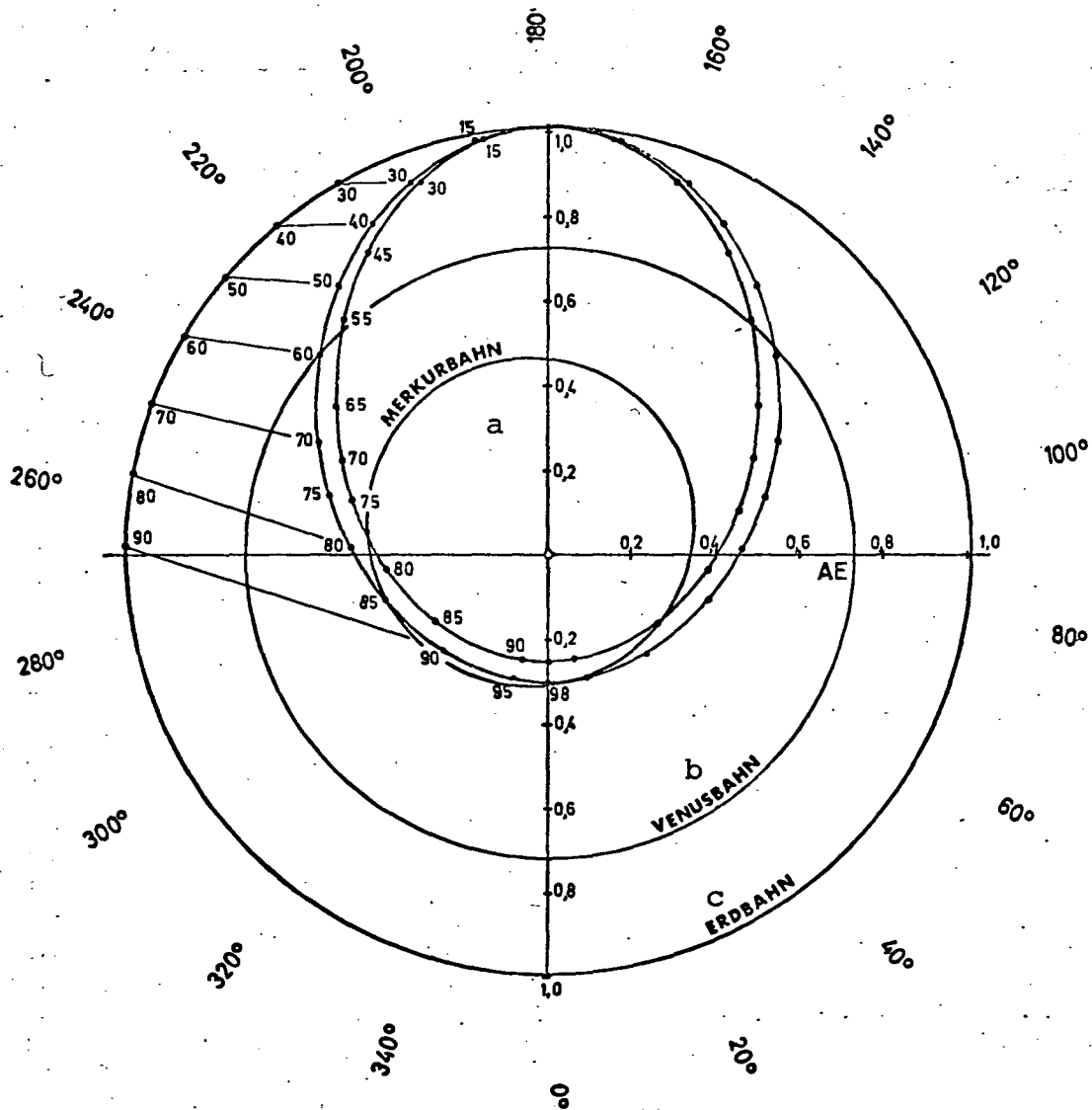
FIGURE 5-02 shows, for example, that an orbit with approximately 0.24 to 0.26 AU possesses the shortest "dead periods" (32) days), whereas an orbit with 0.28 to 0.3 AU has "dead periods" of up to 85 days.

For this reason the choice of an orbit with approximately 0.26 AU is the most favorable; it is also called "half-year orbit", since its pertinent orbit period is half a year.

The second great advantage of the representation of FIGURE 5-02 is the possibility of being able to directly scan the earth-sonde distances. The exact earth-perihelion flight periods as a function of the launching time are represented in FIGURE 5-03. The minimum flight period occurs during maximum required thrust, i.e. the favorable launching times in summer are linked with a flight period to perihelion lasting approximately four days longer, which however is practically insignificant.

The possible disturbances of the free-flight orbits with 0.1 to 0.3 AU perihelion caused by the radiation pressure of the sun and by the planets Venus and Mercury were also investigated. Determining factors in the latter case are the distances sonde-planet, the course of which is represented in figure 5-04.

FIG 5-01



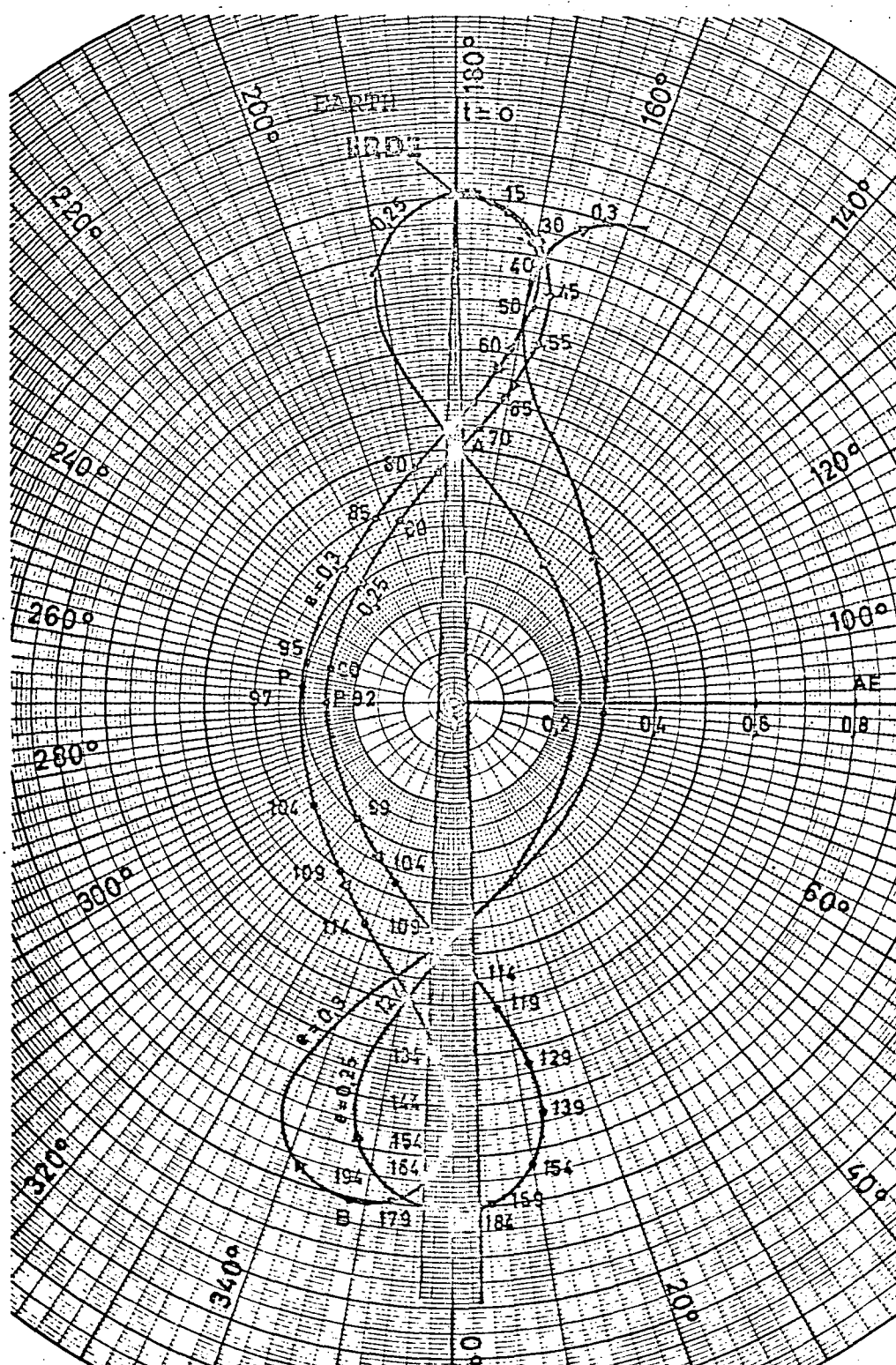
Orbit of a solar sonde with 0.3 and
0.2? AU perihelion in a fixed co-
ordinate system

a = Mercury orbit

b = Venus orbit

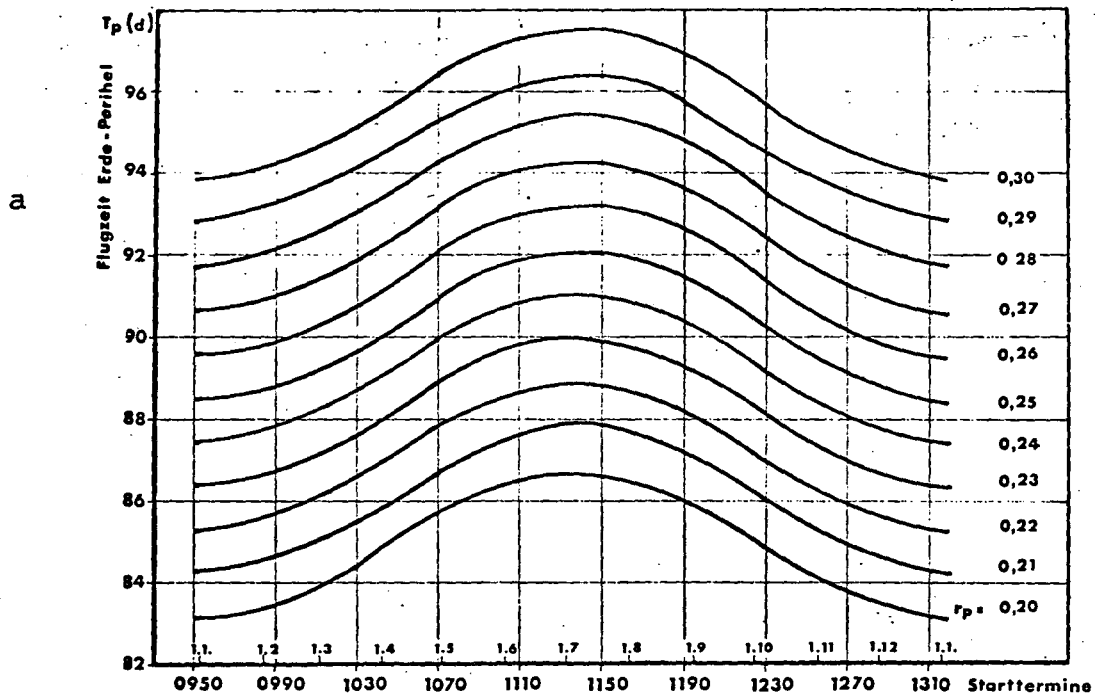
c = Earth orbit

FIG 5=02



Solar-sonde orbits in a rotating
coordinate system

FIB 5-03



b

Flight periods as function of
the launching time.

a = flight period earth - perihelion

b = launching times

According to this, the sonde does not enter the activity sphere of the planets, the radius of which is defined as follows :

$$\begin{array}{lcl} \text{Venus : } r_{\text{act}} & 4.12 \cdot 10^{-4} & \text{AU} \\ \text{Mercury: } r_{\text{act}} & 7.5 \cdot 10^{-5} & \text{AU} \end{array}$$

There is, however, the possibility of getting close to Mercury by small orbit corrections or variation of the injection parameter, if one wants to make use of the (small) orbit disturbances, in order to more accurately determine the mass of Mercury.

5.2 Orbits with Venus Passage

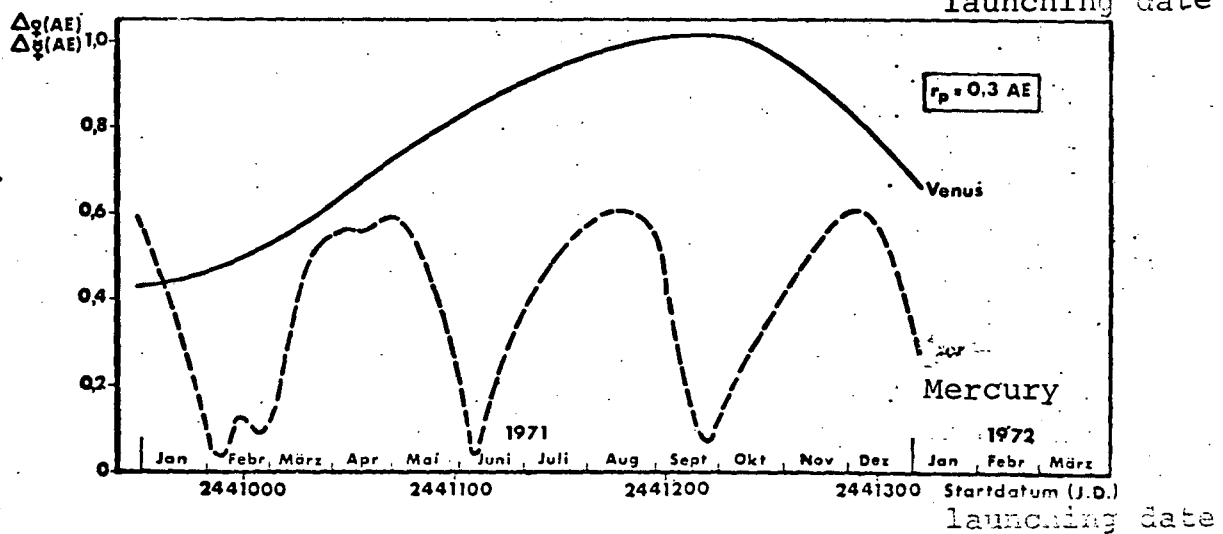
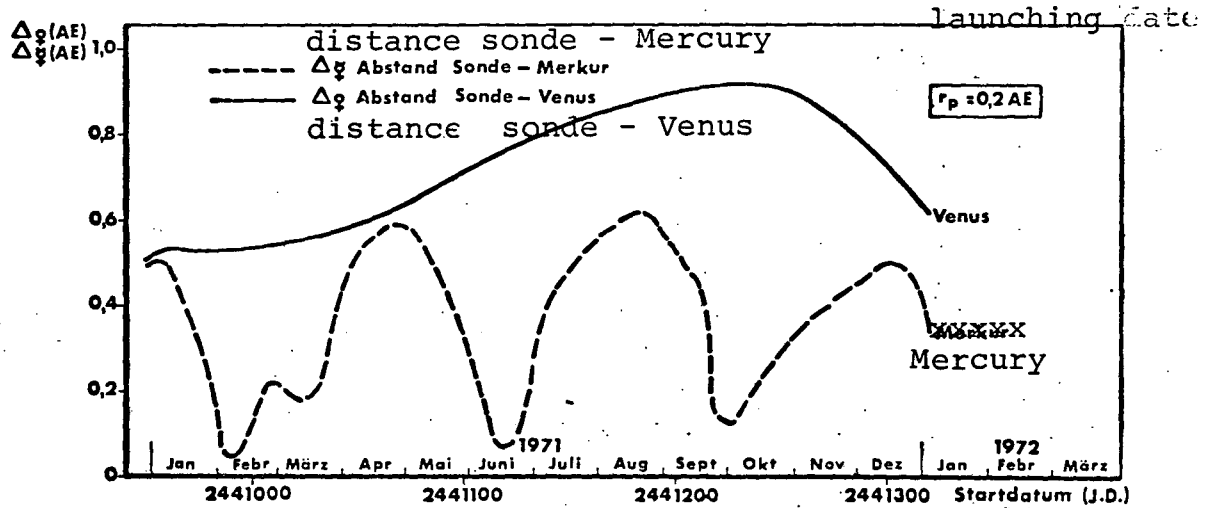
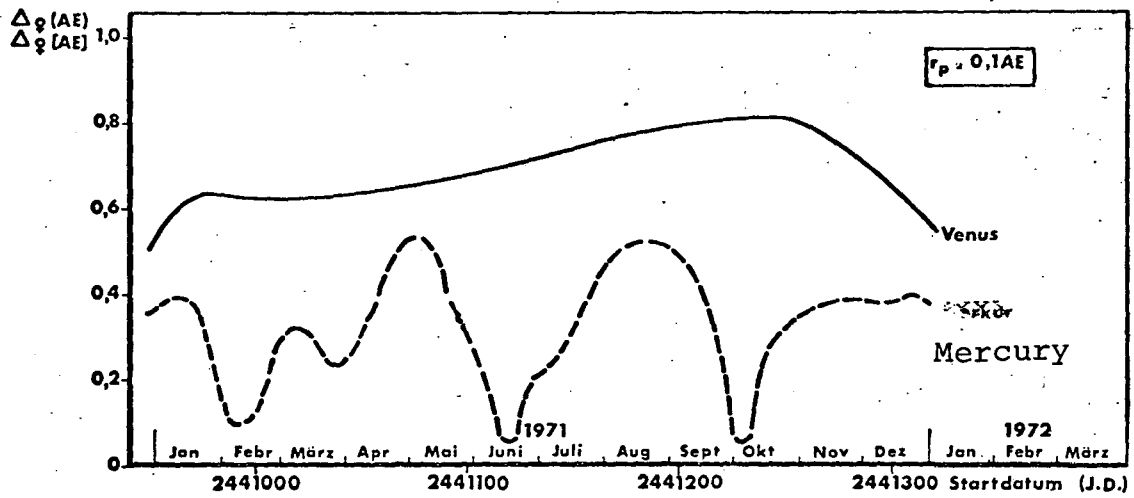
Besides the unintended orbit disturbances, however, it is possible, consciously to make use of this effect in the case of the planet Venus.

The possibility of the Venus swing -by was investigated in detail the results are represented in Report TN-P6B-40/66 .

As an example, a mission was selected which makes possible a Venus passage on 10 May 1972. The advantages with regard to required thrust are shown in FIGURE 5-05.

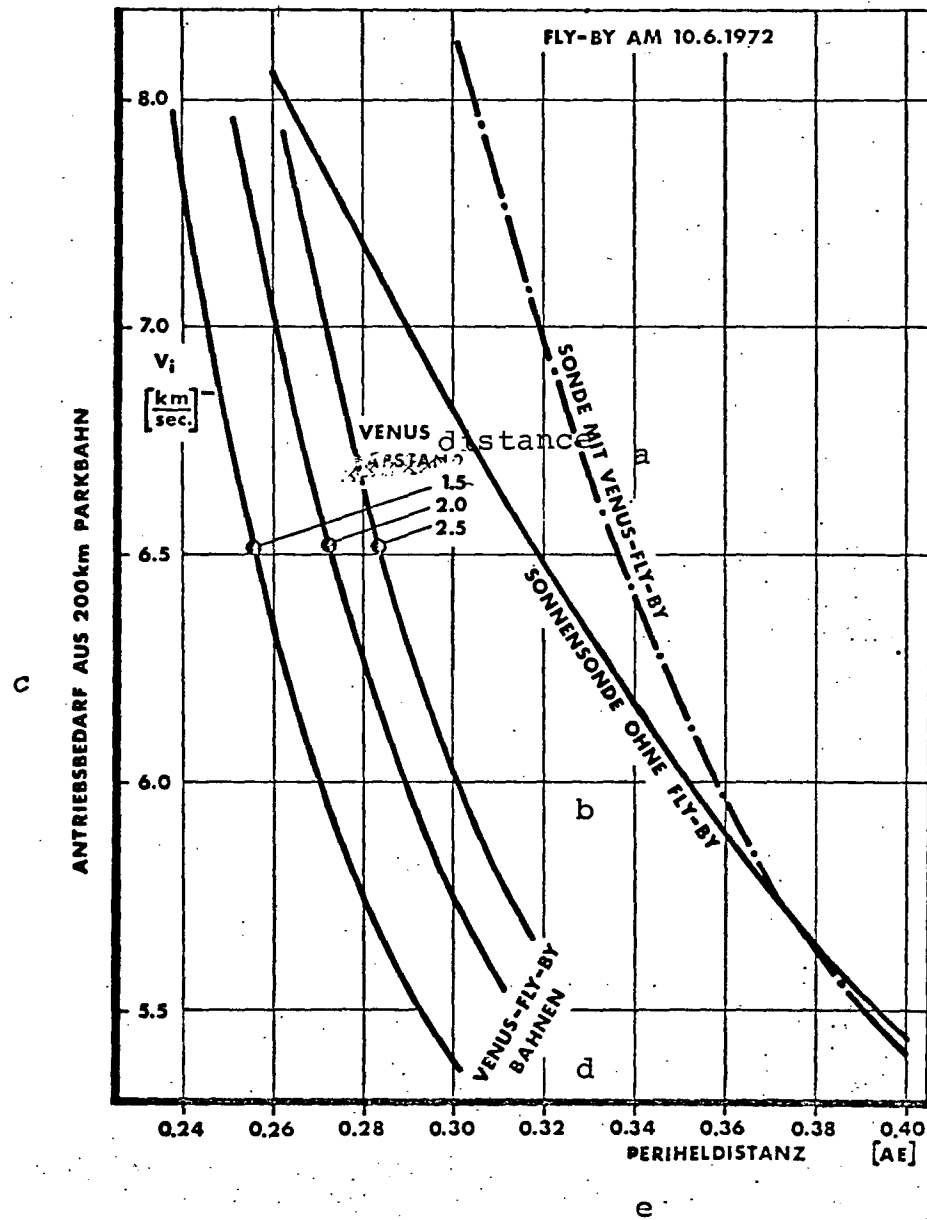
FIG 5-04

-92-



Distance sonde - Venus and Sonde - Mercury for launching times from Jan. 1971 to March 1972.

FIG 5-05



- a= sonde with Venus fly-by
- b= solar sonde without fly-by
- c= required thrust power 200 km parking orbit
- d= Venus fly-by orbits
- e= perihelion distance

Without a swing-by maneuver a ΔV of approximately 6,800 m per sec is required for a perihelion of 0.3 AU (from a parking orbit at 200 km altitude) for a Venus passage at 9300 km distance from the center (1.5 r) or at 3100 km altitude above the Venus surface, the required thrust is, on the other hand, only 5,370 m per sec. Thus 1,430 m per sec are saved or approximately 250 kg useful load are gained (ATLAS-CENTAUR-HetAM), i.e. more than enough to carry out, as a secondary task, experiments concerning the planet Venus. As an alternative, on the other hand, the perihelion distance of 0.4 AU to 0.3 AU or 0.33 AU to 0.25 AU can be diminished by a swing-by maneuver with the same velocity requirement. On the other hand, of course, the mission becomes more complicated due to a Venus passage, and the launching times are narrowed.

5.3 Mission Profile

5.3.1 Launching date and Target orbit

After the general discussion, the orbit analysis shall now be taken up on the basis of an example;

As has already been shown in Chapter 2.2. it is expedient to launch the sonde in summer; in consideration of the required period of development,

3 June 1972

is therefore scheduled. In the preceding chapter it was shown that the half-year orbit with 0.26 AU perihelion offers certain advantages with regard to radio transmission ; moreover, this distance is sufficient for the scientific program. Furthermore, the efficiency limit for ATLAS-CENTAUR HetAM lies at 0.26 AU, and up to this approach to the sun, the technical problems can be managed comparatively well.

These reasons speak for the choice of the 0.26 AU half-year orbit.

The transition period (free-flight phase) is then 93.25 days; the siderial period of the sonde is 186. days and the synodic period is 382.0 days.

Geocentric Phase (Propulsion Phase)

The launching of the sonde shall take place from Cape Kennedy (80° 50' longitude west, 3830' latitude north); the launching azimuth shall be 90°. Thus the plane of the ascent path, which runs directly into the plane of the escape hyperbola, lies at a 2830 - inclination to the equator.

FIGURE 5-06 shows the course of the ascent path for ATLAS-CENTAUR-HetAM to the beginning of the free-flight orbit. In this case injection takes place directly, i.e. without a parking orbit. The results of the orbit, calculated for this special

case, are as follows :

x + 0	sec	launching and subsequent perpendicular ascent
x + 20	sec	beginning of the climbing program with 0.7° /sec up to 70° to the local perpendicular at the launching point.
x + 148.2	sec	propellant cutoff of the booster engines (BECO) at 56.5 km altitude, $v = 2663$ m/sec.
x + 151.3	sec	separation of booster engines with fairing
x + 208	sec	jettisoning of the Centaur insulation and useful load fairing
x + 221	sec	propellant cutoff of the march engine (SECO) at 113 km altitude, $v = 3855$ m/sec.
x + 223	sec	stage separation
x + 230	sec	ignition of the Centaur engines
x + 673	sec	propellant cutoff of the IIInd stage at 183.3 km altitude; $v = 7354$ m/sec.
x + 675	sec	stage separation
x + 680	sec	ignition of the HetAM engines
x + 1322	sec	Propellant cutoff HetAM, altitude 2000 km, $v_b =$ $= 13,820$ m/s ($v_i = 14,261$ m/sec), distance from launching point 8,975 km (81°)
x + 1400	sec	separation of the sonde.

In its first phase, the orbit has been optimized to a 200 km orbit (quasiparking orbit) under the given conditions, a direct optimization with transition into the hyperbola yields a course which leads through the earth.

The total of the yielded propulsion capacity (without consideration of earth rotation) is 16,369 m/sec, and the final mass is 623 kg. Deducting the propellant-cutoff weight of HetAM, there remains a maximum sonde mass of 163 kg.

For determination of the exact launching time, it is necessary to start from the initial heliocentric conditions (3 July 1972, 0 hours world time or 1 hour Central European time). Every 12 hours there is a possibility of passing from the launching-path plane into the heliocentric plane.

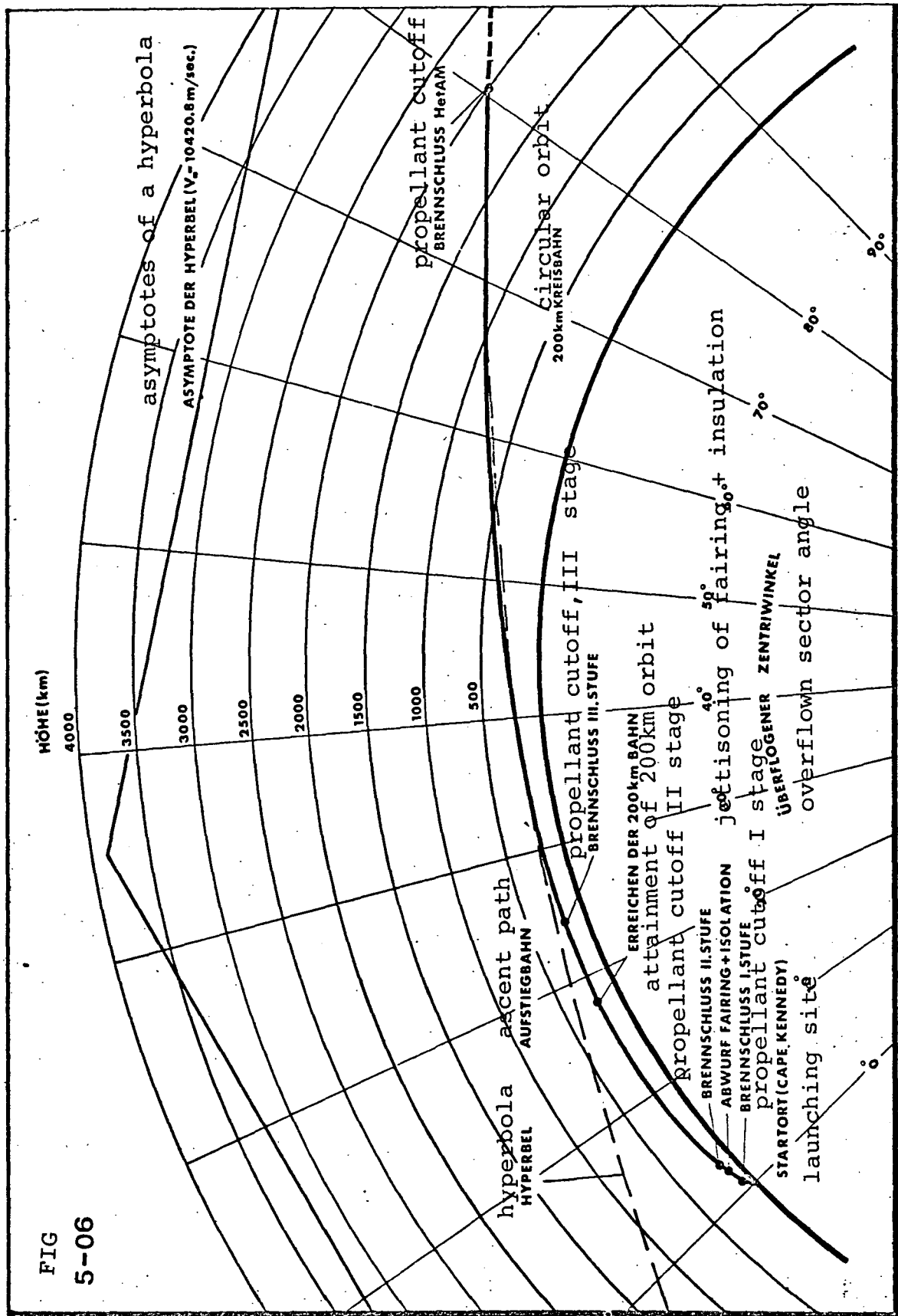
In consideration of the effective ascent time to the point of injection, the following launching times are obtained:

1 July 1972	16 ^h 56 ^m	WT
	4 ^h 54 ^m	WT
	5 ^h 54 ^m	WT
	17 ^h 52.5 ^m	WT

Altitude

FIG

5-06



Course of the ascent path.

By variation of the launching azimuth (in the permissible range) the launch windows can be somewhat varied.

In summer, the difference between world time and Eastern Standard Time (Florida) is 6 hrs.

5.33 Heliocentric Phase (Free-Flight Phase)

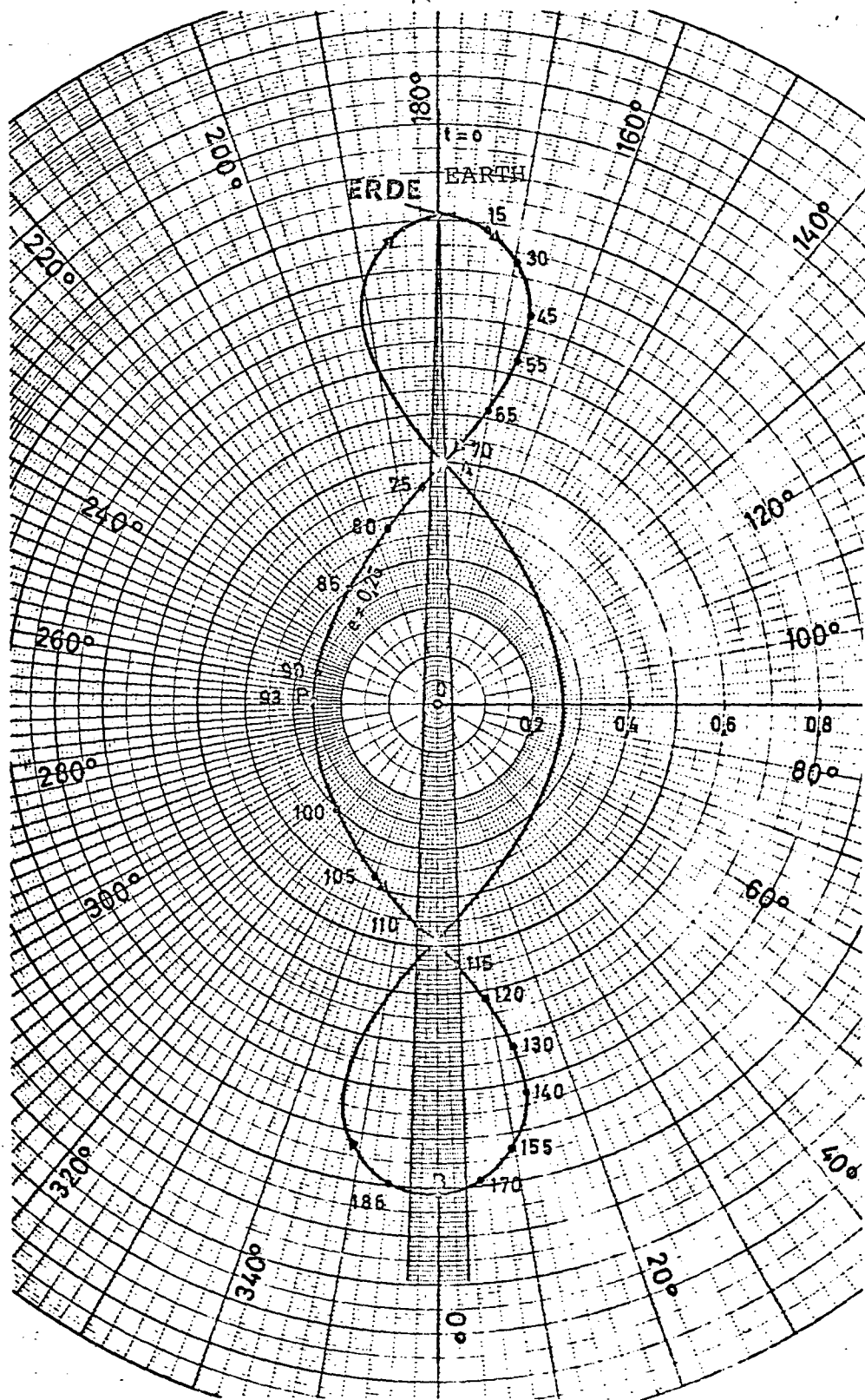
The selected orbit with 0.26 AU perihelion is represented in FIGU. 5-07. We are dealing here with a representation with a rotating coordinate system, the advantages of which have already been mentioned.

The "black-out" zone for data transmission was drawn in assuming an angle of 160 arc minutes, which takes into account the higher layers of the K corona.

The first disturbance of the data transmission occurs approximately 72 days after launching, it lasts about 2 days. The second passage through the disturbance zone occurs after 110 days and lasts about 6 days, while the third and last passage begins after 175 days and lasts approximately 28 days.

The earth-sonde distances may be taken directly from the diagram for each point in time.

FIG 5-07



Half-year orbit in a rotating coordinate system

As has already been shown in Chapter 5.2, no disturbances are to be expected from Venus or Mercury for the launching time under consideration.

There is, however, a possibility of coming relatively close to Mercury' if the launching took place on 20 August 1972, the sonde would approach so close to Mercury so far (passage on 3 November 1973) that, due to the insignificant orbit disturbance connected therewith, the mass of the planet Mercury could be determined considerably more closely than hitherto (the uncertainty is today still approximately 20 per cent). This could become an interesting secondary task of the mission.

- 6. DESIGN CRITERIA OF A SOLAR SONDE
- 6.1 Navigation and Position-Regulation System
- 6.11 Requirements with regard to position stabilization

As has been shown in Chapter 2.13, a position-stabilized sonde is required for carrying out the experiments.

The experiments also determine the navigation and measurement accuracy of 4 arc minutes, but not the actual position-regulation accuracy. The latter is determined by technical parameters, namely by the construction of the conical tubus as well as by the temperature control of the solar-cell surfaces. Both require a position-regulation accuracy of approximately $\pm 2^\circ$ thus no particularly critical requirement.

The system can be constructed relatively simply, since the conditions for stabilization during the entire flight phase are equal. No shadow periods and no mid-course maneuvers occur, which normally also require, in addition, a platform as a reference system.

In the case of a solar sonde it suffices to employ one sensor system as a basis for reference. As suitable reference points there are the sun and Canopus. The main axis of the sonde must be directed to the sun for reasons of measurement technique.

If a planet, for example the earth, is taken as a second reference point, such a constellation may occur that sun, sonde and planet lie on one axis, and the sonde is no longer stable. If, however, the two axis stand almost perpendicularly upon each other, the two system axes are independent of each other.

Therefore, it is expedient to use sun and Canopus as reference points.

6.12 Structure of the navigation system

In order to find the sun after launching, a crude sensor with a dome-shaped field of vision is used (2π sterad). If the position of the sonde is reached to within $\pm 15^\circ$, an optical switch switches over to a fine sensor, which possesses a field of vision of $\pm 15^\circ$. It generates an almost linear signal over the entire region, and makes it possible to adjust the sonde within a range of $\pm 2^\circ$.

The Canopus - sun axis stands almost perpendicularly to the ecliptic (deviation $\pm 15^\circ$). During one orbit around the sun, Canopus thus moves along a cone with an aperture angle of 30° . Nevertheless, the star sensor is installed perpendicularly to the solar-sonde axis, since the field of vision of the

sensor can be varied and adapted electronically.

During the position search, the star sensor must be protected by means of a cutoff against accidental solar radiation. In addition, one or two earth sensors are installed, which are required in order to adjust the antenna. As "back up" they serve at the same time for the star sensor, in case it should break down. The behavior of the individual components will be described later.

TABLE 6-1 provides a compilation of the system data.

The total volume is approximately 20 dm^3 . The total reliability of the position stabilization (without moment transmitters) is 0.955.

6.13 Compilation of Component Data

(a) The crude sun sensor (sun searcher)

The crude sun sensor has the task of roughly directing the sonde, moving freely in space, toward the sun. Due to the albedo and other disturbances, such a sensor can adjust the sonde only to approximately $\pm 4^\circ$. Therefore a fine sensor, with a field of vision of 15° , is needed in addition.

	1	Abmessungen (mm)	2	Gewicht (kg)
a	Grobsensor	80 x 80 x 20		0,08
b	Feinsensor	60 x 60 x 40		0,12
c	Umschalter	25 x 25 x 80		0,03
d	Sternsensor	100 x 140 x 300		3,50
e	2 Erdsensoren	je 100 x 140 x 300		6,40
f	Elektronik	200 x 300 x 105		3,00
				13,13 kg

TABLE 6-1: Weights and Dimensions
of the Sensors.

a = crude sensor

b = fine sensor

c = switch

d = star sensor

e = 2 earth sensors

f = electronic equipment

1 = measurements

2 = weight

In principle, the crude sensor consists of 4 solar cells, which are mounted perpendicularly in a square upon a small blackened platform.

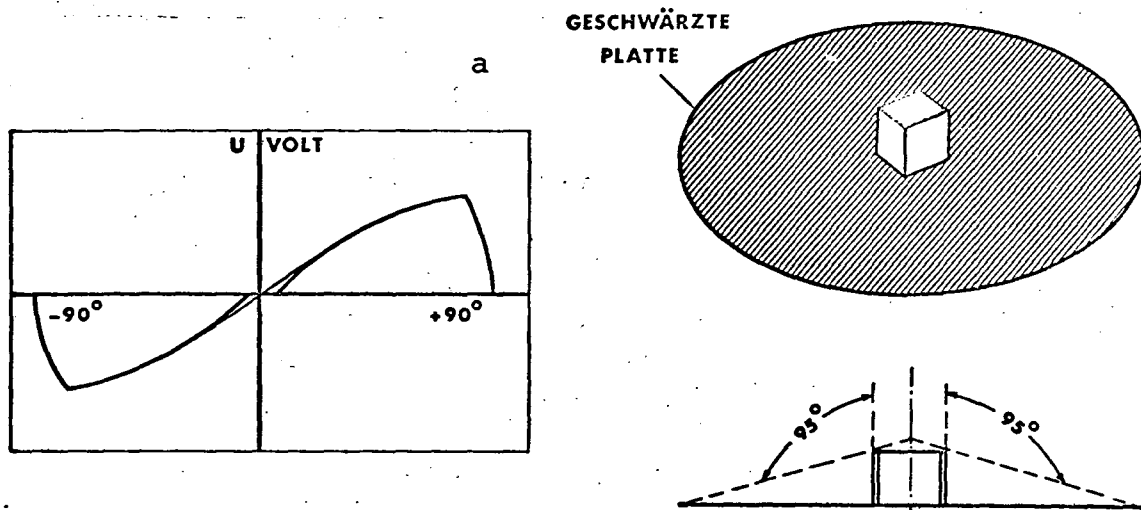
The cells always perceive the sun, as long as they are not hidden behind the sonde. The field of vision is limited by the beam which must be drawn from the upper rim of the cells over the platform edge. It lies at approximately $\pm 93^\circ$. The cells facing each other are coupled against one another. A sine-shaped signal is obtained for pitch and yaw axis.

The construction is represented in FIGURE 6-01, also the measurement characteristics.

The reliability is 0.9995 for one year, assuming a mean life span of 10^8 hours for each cell and the two load resistances.

The crude sensor is employed only near earth, since it can scarcely be assumed that the fine sensor will lose the sun again. In case the fine sensor fails, the rough sensor serves as a reserve (with reduced accuracy).

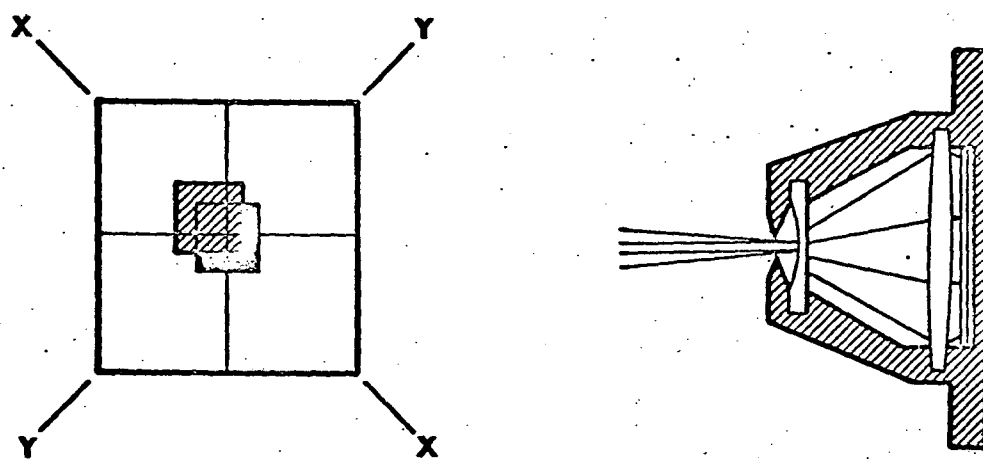
FIG 6-01



Scheme and output signal of the
crude sun sensor (sun searcher) '
Accuracy $\pm 6^\circ$.

a = blacked disk.

FIG 6-02



Design and operation scheme
of the fine sun sensor
Accuracy $\pm 0.15^\circ$.

(b) The fine sensor

The deviation from the ideal position is continuously monitored by means of the fine sensor. Since during flight the incident energy is raised by a factor of 10, the error voltage must be put in a ratio with a comparison voltage, in order to obtain a time-constant signal.

This wide range can be hardly taken care of with a sensor, since the solar cells become too hot. A strong filter must be provided. In spite thereof, this possibility will not be sufficient. It is better to choose a suitable optical system, which magnifies the almost parallel incident beams into a wide parallel beam. The intensity at the cell is thereby reduced. By means of an appropriate design, it is possible to cover the region from 1 AU to 0.33 AU. Near the earth, the cells would then reach a temperature -50° and at 0.33 AU would reach a temperature of 125° C. For reasons of safety two ranges, i.e. two sensors, must be used. The optical system would consist of an aperture, a concave lens, and a convex lens (compare FIG. 6.-02).

The greater one makes the ratio of the intensities, the smaller becomes the possible range of vision of the sensor and all the more do constructional deficiencies make their appearance in the system.

If glancing incidence upon the solar cells is to be avoided, the permissible angle is approximately $\pm 60^\circ$. At an intensity ratio of 4 the vision-field angle of the sensor becomes $\pm 15^\circ$. Since the crude sensor is accurate only to $\pm 6^\circ$ the permissible angle of the vision field cannot become smaller than $\pm 10^\circ$.

Thus the permissible intensity ratio is approximately 6. A normally exposed cell can be used up to a shortest distance of 0.65 AU. If a lense system with an intensity ratio of 6 is then used, up to 0.27 AU is attained. A small improvement can also be obtained through filters. A sensor type, as previously sketched, is thus suited for use in a solar sonde. The comparison voltage must be generated with a similar system; here, however, only a constantly exposed cell is employed. The sensor itself consists of four solar cells which are coupled against each other.

(c) The switch

The switch has the task of switching from the crude sensor to the fine sensor. As has been mentioned previously, this takes place at $\pm 10^\circ$. A simple method is to narrow the vision field of a photodiode to such an extent that it gives a signal only in a very narrow angle range. The diode must be protected from overload by a filter.

(d) The Canopus sensor

Due to the width of 70° with respect to the ecliptic, the sensor must possess a field of vision of $\pm 18^\circ$, since the sonde moves in an orbit around the sun. Due to the long flight period, sensors with movable parts are excluded for reliability reasons. Hitherto sensors were built by JPL and ITT. They are comparable in their performance. TABLE 6-11 gives some of the most important data for the ITT sensor.

The sensor must be protected from strong incidence of light.

During the search for the sun, the sensor must be made insensitive by means of a cutoff.

TABLE 6-II

Chief data of the ITT Canopus Sensor

Field of vision:	18° x 8.2° can be electronically extended to 15°.
Accuracy:	+/- 3 min
Signal:	1 V/degree linear up to +/- 2°
Canopus	
"capture"	angle velocity 2°/ sec.
Magnetic-field sensitivity	0.33 sec/Gauss
Generated magnetic field	5 x 10 ⁻³ Gauss at 10 cm distance
Weight	3.5 kg
Rate of power input	
Dimensions	10x14x30 cm
Reliability	0.95 for 1 year

(e) The earth sensor

In order to trail the directional antenna, an earth sensor is needed. It is employed only at approximately 0.5 AU, since before that a polydirectional antenna is sufficient. At this distance the earth has the effect of a very bright star. Therefore a star sensor of the ITT type, which however, does not have to be so sensitive, can be employed here, too.

A difficulty arises, however, due to the fact that the radiation intensity continuously decreases with increasing distance, and with it also the gradient of the output signal. Since, however, the sensor is only employed for orientation of the antenna on the axis and the deviation angles need not be transmitted to earth, this disadvantage is not aggravating.

But if a constant signal is necessary, a comparison signal must be generated or a correction concerning the calculation of distance must be carried out; for the brightness decreases with the square of the distance from earth. The sensitivity must be continuously increased during the flight. If the Canopus sensor should fail, the earth sensor can also be employed for stabilization of the roll axis. This possibility however loses accuracy the smaller the angle between sun-sonde and earth-sonde becomes.

(f) Position-regulation electronic equipment

An accurate design of the electronic equipment is not possible. However, some statements can be made on the basis of data concerning similar systems.

The electronic equipment can be constructed in three types, with

- a - individual components
- b - individual components and integrated switching circuit
- c - individual components and mechanical relays.

If the conventional construction with individual components (compact (gedr.) switches) is put equal to one, the following relative relationships are yielded :

	a	b	c
Individual components	1	0,14	0,95
Integrated weitch circuits	-	0,075	-
Relays	-	-	0,012
Reliability for 1 year	0,74	0,88	0,16
Weight (kg)	1	0,38	1,33
Specific volume (dm) (kg)	2,1	3,0	2,54
Volume (dm)	1	0,54	1,59

The values cited above refer to a system employing two different reference systems, which are continuously switched about. It must therefore be assumed that in the case of the solar sonde the total outlay for electronic equipment will become smaller.

A reduction by twenty-five per cent is assumed. Then for a and b the following values are obtained:

	a	b
Individual components	3250	300
Integrated switch circuits	-	180
Weight (kg)	3,0	1,2
Volume (ccm)	6300	3400
Reliability for 0,5 year.	0,89	0,953

The advantages of the employment of integrated switch circuits are obvious. Nevertheless, the normal construction with individual components will for the time be assumed (dimensions: 20 x 30 x 10.5 cm).

6.14 Search for Position

After the separation of the sonde, a moment is exerted upon the pitch or yaw axis. The sonde rotates until the sun moves into the field of vision of the crude sensor. The crude sensor now automatically takes over the adjustment of the sonde.

Below the 1° limit the change-over switch gives a signal whereupon the fine sensor assumes the exact adjustment to $\pm 2^{\circ}$. If the longitudinal axis is oriented to the sun, a signal is given from the earth which gives a roll impulse. In a lateral fluctuation range of $\pm 8^{\circ}$, the star sensor searches for Canopus. The sensor reacts only to a certain range of brightness. Since in the section of the sky seen by it, there is at this moment only one star of the brightness of Canopus, it will react to it and will again retard the rotary motion of the sonde.

FIGURE 6-03 shows an example of a star map. A comparison between the measured and the predicted curve is interesting. All of the measured values are higher. It can thus also be explained that the star sensor of Mariner IV several times took bearings on another star (for instance Regulus). For each launching time the star map must be set up anew. During one revolution the sensor sees a band of $\pm 8^{\circ}$ which is different according to each launching date.

6.15 Initial conditions and perturbation moments for the design of the adjustment system

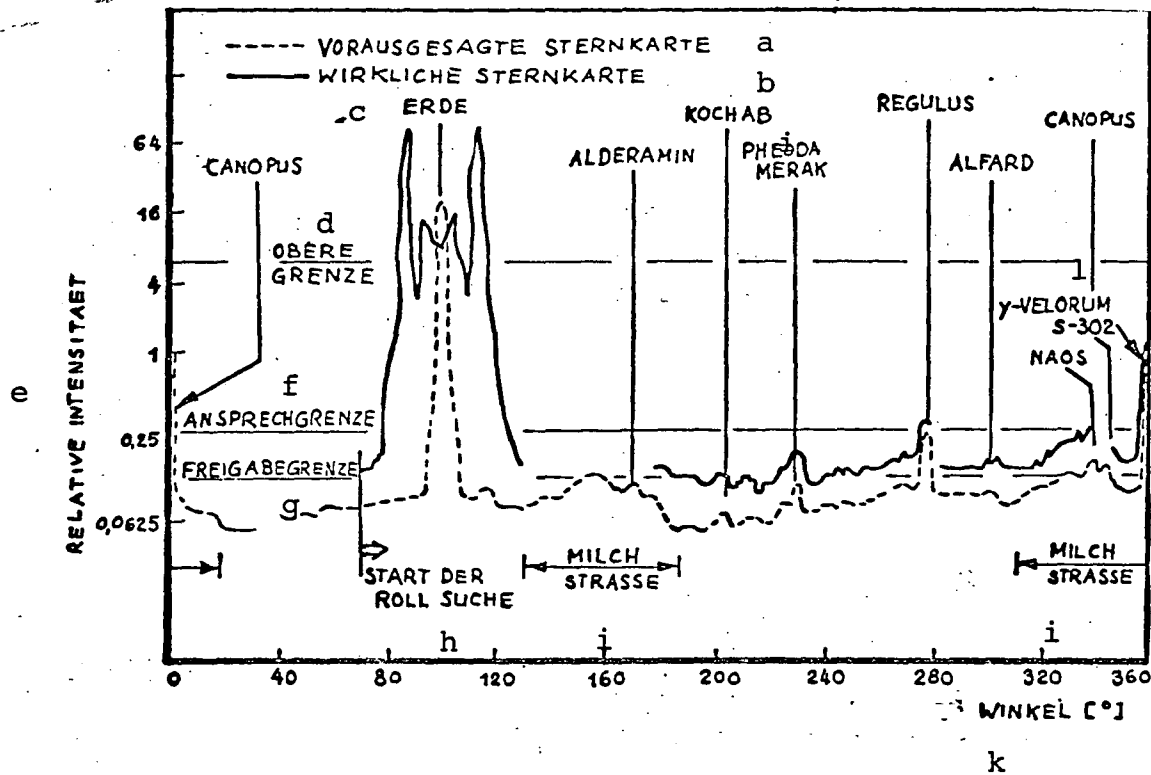
There exists a whole series of possibilities for equalizing perturbation moments. Since the total weight of the device is relatively small, most of the torque inducers are out of the question.

The angle deviations in the case of the solar sonde are rather large and the perturbation moments are small, so that the required gas weight is chiefly determined by the amount of leakage gas. Position-regulation systems with cold gas (nitrogen) have been tested so often that they are well known, and can be adapted to all requirements. Their reliability is very well known. It is this very point that is of great significance.

Three measurement-instrument packages for the solar sonde, which make varying demands upon the degree of position stability, are under investigation. If the actual instantaneous position angle is also measured, a maximum deviation angle of up to 10° can be permitted for the "minimum package".

Another limit is formed by the aperture angles of the antennas which decrease constantly as the bit rate increases. All restrictions, however, still lie within the range of several degrees.

FIG 6-03



Star map of the Canopus sensor
for stabilization of the sonde
around the roll axis.

- a= predicted star map
- b= real star map
- c= earth
- d= upper limit
- e= relative intensity
- f= sensitivity limit
- g= release limit
- h= start of the roll search
- i= Milky Way
- k= angle

It is unlikely that adjustments smaller than 2° will become necessary. Since it will later be seen that the gas weight is very small, further calculation will be based upon this assumption.

Perturbations are caused by

- solar pressure
- leakage gas
- radiated energy
- meteoroid hits
- change of angular momentum in the sonde
- gravitation fields
- magnetic fields.

Almost all of these perturbations are small in comparison to solar pressure and leakage gas. In the case of very rapidly accumulating magnetic tape storage, a noticeable perturbation can also come about due to change of the angular momentum.

All these influences are, however, not decisive for the dimensioning of the cold-gas system. It is very difficult to calculate the perturbation due to solar pressure with sufficient accuracy.

A surface 1 m^2 in size on a 1 cm lever arm undergoes, for

example, near earth a torque of approximately $M = 1.4 \times 10^{-5}$ mp.)

These values, as well as the other perturbation moments due to leakage rates, are by several powers of ten smaller than the produced momentum (3 mp). The design of the gas-nozzle system is determined only by the technological limits.

6.16 Calculation of the required amounts of gas

The smallest of the tested gas nozzles possess thrusts of a few pondes. The problems of manufacturing still smaller nozzles have to do with the very small openings which are necessary to regulate the gas flow. Due to the relatively inaccurate processing, the efficiencies decrease, and with them the specific impulses. The lowermost thrust limit lies at approximately 1 p. In order to avoid unnecessarily great difficulties in manufacture from the very beginning, a thrust of F 3 ponds is assumed for further calculation. The specific impulse for nitrogen gas at 300° K is $I_{sp} = 60$ sec. With these values the calculated drive is 0.05 p/sec.

The sonde moves back and forth with the limit cycles. This limit cycle is determined by the smallest possible impulse. The smallest opening and closing time is five milliseconds. A pure flow time of 10 m sec. is planned in order to catch the sum of dead periods of the sensors and of the electronic

equipment. Cycle time $T = 7760$ sec is obtained with a deviation angle $\beta = \pm 2^\circ$, $\theta = 2.5 \text{ kp m sec}^2$, $l = 1 \text{ m}$ and $I = 45 \times 10^{-3} \text{ p sec}$.

The gas supply for one year flight period (approximately one solar orbit) amounts to a total of 9.2 grams at 4060 cycles for all three axes. In addition to this gas quantity the leakage gas must be included in the calculation. The leakage rate is 0.5 ccm/h. In the case of 6 nozzles a leakage volume of approximately 26,300 ccm is yielded in one year, which corresponds to a weight of approximately 33.7 p. The required total weight is thus altogether 42.9 g.

Since the maximum leakage rate will probably not occur, there is still a reserve for actual position regulation. In spite of this, since there are still too many unclarified parameters in the calculation, a gas weight of 100 p is to be reckoned with..

Attention must be directed to another point; in this calculation the separation of the sonde and the search of the angle position were not yet taken into consideration. It is possible that the initial angle velocities are very high, and that a larger momentum is required for the deceleration of the sonde. From comparison of existing sondes and investigations it is unlikely that this case will occur; in spite of this a

reserve for it shall be provided for, so that the total gas amount of 200 g is entered in the calculation.

6.17 The gas-nozzle system and its components

The construction of the system and the components belonging to it can be seen in FIGURE 6-04.

The individual components were investigated in detail and described in Report TN-P6B-38/66. Since they are simple and conventional parts, they will not be treated here in detail. The weight and reliability values (for one-year operation) are compiled in TABLE 6-III

	Weight (g)	Breakdown rate
1 N ₂ gas container (0,6 l, Alu., 300 ata)	960	0,00035
N ₂ - Gas	200	-
6 solenoid valves with nozzles (KIDDE)	420	0,07295
1 explosive valve (NA)	180	0,0026
1 pressure-reduction valve (STERER)	450	0,00313
2 Filter	200	0,000348
1 fill-up and drain valve	65	0,00087
1 pressure meter	100	0,06080
1 temperature feeler	20	-
lines, connections, etc.		
including 10 per cent safety factor	480	-
TOTAL WEIGHT	3085 g	

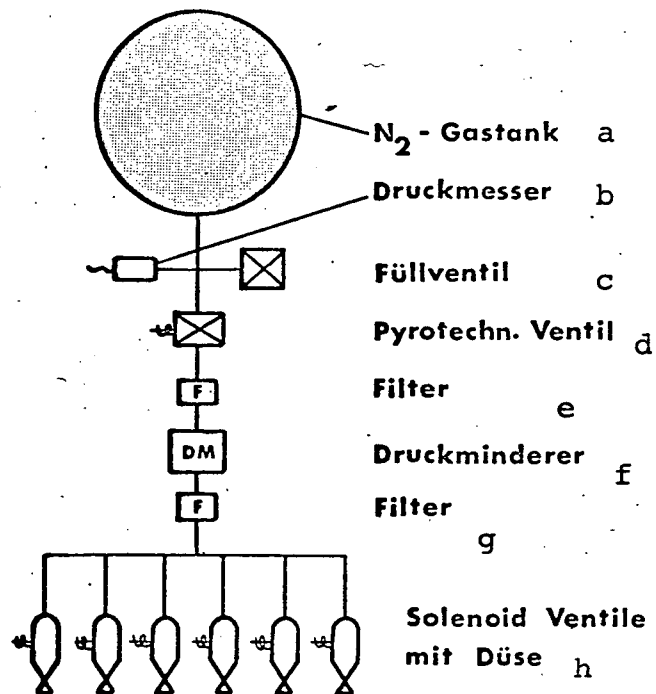
With these values a total reliability of the system of 0.837 is yielded. This value of 83.7 per cent can of course still be improved by redundancy.

If a second pressure meter, a second set of solenoid valves and two additional pyrotechnical valves are installed, a system of 91 percent reliability is yielded. The weight would be increased by 0.88 kg to 3.965 kg.

Another set of valves would bring the system up to 93.1 per cent. Additional more extensive improvements are possible only with considerable difficulty. It must, however, be kept in mind that the system keeps on functioning even if the pressure meter breaks down, so that the actual reliability is higher.

This means that the initial reliability increases to 92 per cent. With two sets of valves, as much as 95.5 per cent is obtained, a value which must by all means be aimed at. The additional leakage-gas weight is still within the permissible limits of the gas and tank weight.

FIG 6-04



Scheme of the gas-nozzle
system for position regulation.

- a = N₂ gas tank
- b = pressure meter
- c = fill-up valve
- d = pyrotechn. valve
- e = filter
- f = pressure reducer
- g = filter
- h = solenoid valves with nozzle

6.18 Overall Design and System reliability

Of the navigation system, the rough sensor and the switch are practically only once briefly engaged. Their reliabilities must then be set practically equal to one. All other components are necessary for successful position stabilization.

The earth sensors are only needed for orientation of the antennas. They belong to the data transmission.

Solar sensor	0,9999	for 0,5	years
Star sensor	0,975	"	"
Electronic equipment	0,89	"	"
Total reliability	0,867	"	"

This value is almost sufficient without redundance. The momentum giver, i.e. the gas-nozzle system, has a function reliability of 95,5%, as was shown in Chapter 6.17.

Thus a total reliability of 82.8 per cent is reached for longitudinal stability during 1/2 year (up to the perihelion of the orbit).

This brief treatment has shown that a position-stabilization system for a solar sonde can be built without great difficulty. To a considerable extent, components already built can be used. These, however, were built almost exclusively in

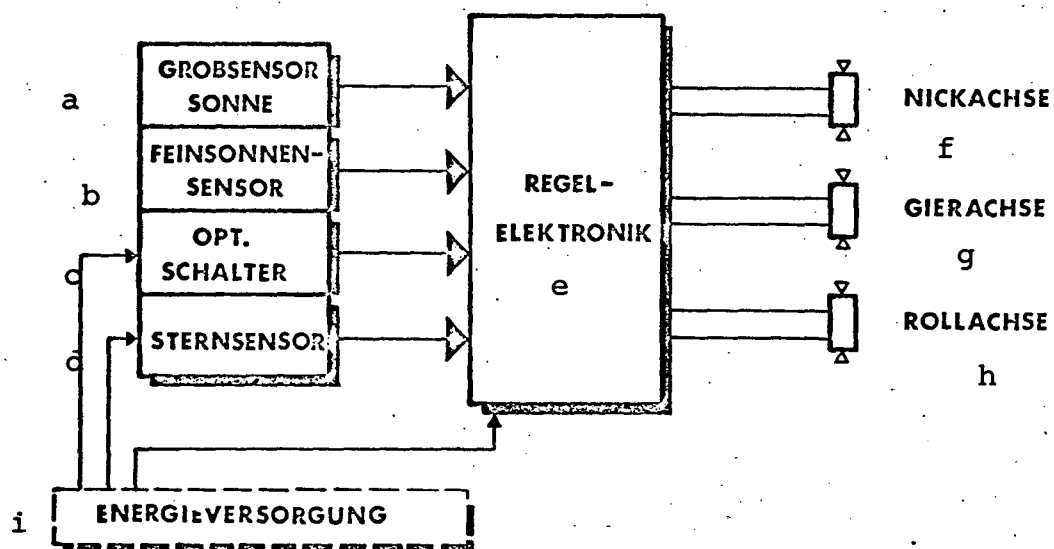
the USA, so that on occasion they would have to be built under license.

The reliability is sufficient. The required position accuracy was assumed at $\pm 2^\circ$ about all axes. In the case of exact designing, it should be possible to complete the system within a margin of a few (approximately 20) arc minutes. By the development of special fine sensors, the position accuracy can be still increased. However, the system as here designed is sufficient for the proposed experiments even in the case of the maximum mission. Other restrictions due, to the required position accuracy of the antenna, the solar-cell arm and the heat shield, are not so aggravating.

The weight of the device is low, and can be reduced still further by the employment of other earth sensors and integrated switching circuits, at the same time increasing the reliability still more. Detailed individual investigations can lead to still more efficient systems. However, the investigation conducted here is already sufficient for carrying out a solar-sonde mission.

FIG 6-05

SCHEMA DER LAGEREGLUNG
SCHEME OF THE POSITION REGULATION



- a= crude sensor, sun
- b= fine sensor
- c= opt. switch
- d= star sensor
- e= electronic adjustment equipment
- f= pitch axis
- g= yaw axis
- h= roll axis
- i= energy supply

6.2 Data Transmission

The data-transmission system has the task of transmitting scientific measurement results as well as function-technical data from the sonde to earth. The required transmitting capacity depends upon the information flow, distance, the noise level and upon the transmitter and receiver installation.

The compilation of the information flow is contained in TN-P6B-39/66 and 44/66. Due to appropriate choice of the measurement cycles, the scientific information flow could remain limited to 12.5 bits/sec. The function-technical values, which include temperature, pressure, current, and voltage measurements, amount to approximately 8.5 bits/sec. These values are only good for the minimum version of the design.

The useful-load package with a weight of 32 kg represents an extension of the experiments concerning the minimum version.

The maximum information flow of the scientific useful load is thus increased to 472 bits/sec.

It is divided into :

the plasma experiment with	10 bit/sec
the magnetic-field experiment	50 bit/sec
the cosmic-ray experiment	11 bit/sec
the zodiacal-light experiment	1 bit/sec
the micrometeoroid experiment	400 bit/sec

The values for plasma and zodiacal-light and cosmic-ray experiments could not be compressed any further. In the case of the magnetic-field experiment, the same measurement values are basically yielded as in the case of the minimum version (1.7 bits/sec). In case of the measurement of shock waves and plasma disturbances, up to $2 \cdot 10^7$ bits must be processed in a period of 60 sec. This requires a storage accumulator with a capacity of 4×10^7 bits and a read-off rate of approximately 350 bits/sec.

By coupling high-speed digital-differential analysers in front, the read-off velocity and accumulator capacity can be diminished. Another possibility of data compression is the subsequent coupling of a DDA or of an evaluation computer for reading off from the accumulator for data transmission. In each case, from measurements according to the above-mentioned cycle, information-flow values of approximately fifty bits/sec are yielded during a period of one or several days.

By means of suitable choice of the measurement cycle, and the

installation of a tape accumulator of at least 400 000 bits capacity, the information flow can be reduced to 10 bits/sec.

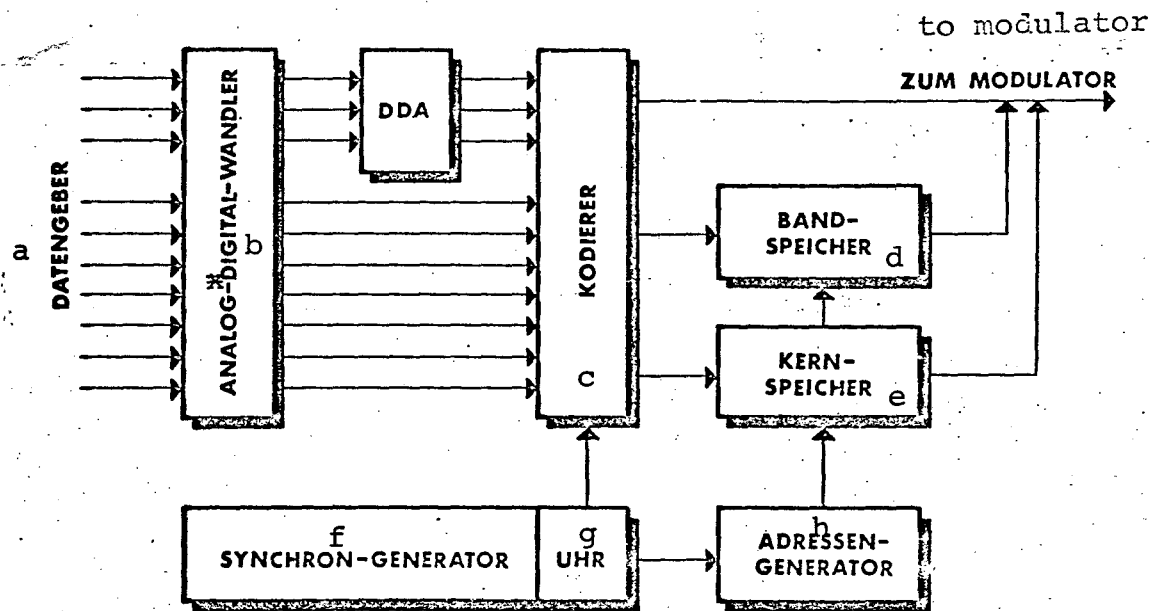
Thus a value of 82 bits/sec is yielded for the maximum information flow, which, except in the case of peak times, can be reduced by a small data yield to 32 bits/sec.

A block circuit diagram of the data processing is represented in FIGURE 6-06.

Only the DSIF system that shall be changed over to 70 m Ø antennas by 1967 can presumably be used as a receiver device. By means of employment of helium-cooled masers, the receiver noise temperature could be kept down to only fifty degrees K. Together with an antenna gain of 61 db, a device is obtained the efficiency of which can be hardly exceeded.

The greatest transmission distance is at 2 AU. Up to 0.5 AU a polydirectional antenna is still sufficient. Then a switch is made to an antenna with approximately 11 db gain. In the case of a transmission capacity of 10 w, the yielded information flow of 22 bits/sec (minimum version) can be transmitted without disturbances, as long as the sonde is not in either the upper or lower conjunction with the sun.

FIG 6-06



Scheme of Data Processing

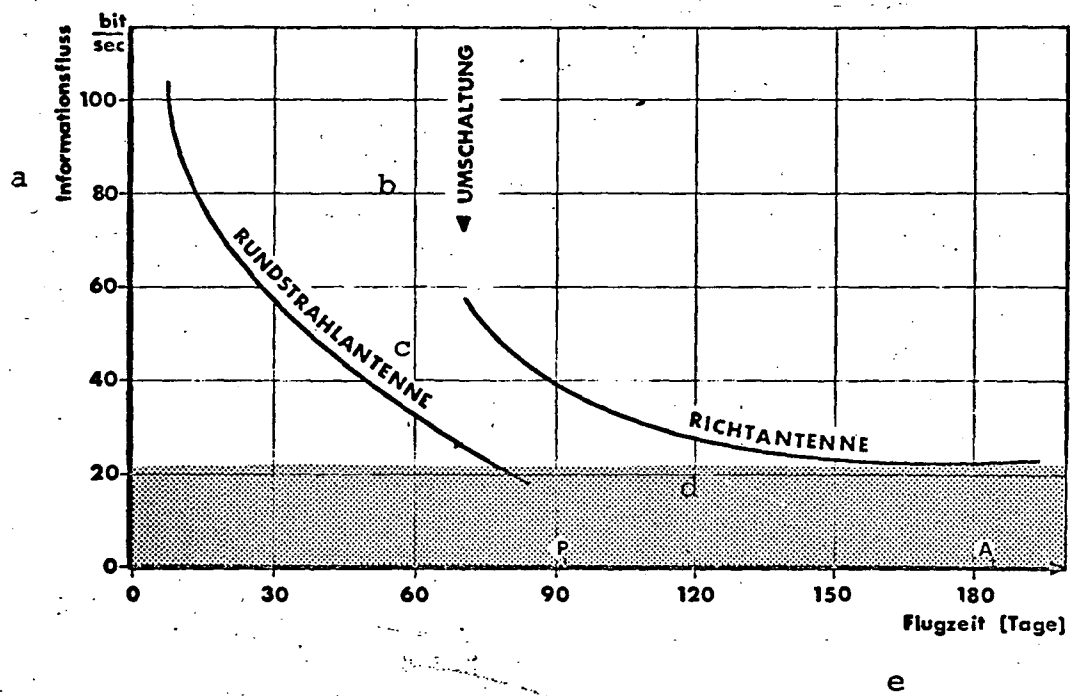
a= data giver
 b= analogue digital converter
 c= coder
 d= band accumulator
 e= nuclear accumulator
 f= synchronous generator
 g= clock
 h= address generator

During times of increased solar activity, the noise temperature of the sun of $5 \cdot 10^4$ ° K increases to a maximum of 2×10^5 ° K in the S band (2,3 GHz). Thus the noise temperature of the reception antenna is increased in accordance with the position of the sun, and the information flow is restricted. The solar eruptions are, however, limited according to intensity with regard to number and duration, so that a noise temperature lying between $5 \cdot 10^4$ and $2 \cdot 10^5$ ° K, i.e. at approximately 10^5 K, can be used as a basis for the design. Detailed information can be found in TN-P6B-47 / 66.

With a mean reception-antenna noise temperature of 300° K, transmission of 22 bits/sec. is possible at an ideal band width of 80 cps. In the case of distances smaller than 2 AU, the possible information flow is increased in accordance with FIG. 6-07, which refers to a half-year orbit. After approximately 70 days a switch is made from the polydirectional antenna to the directional antenna, which, upon command from the earth, is gradually tracked ($\Delta \alpha = 2^\circ$).

A spiral antenna with spherical polarization is employed, which for 11 db gain possesses only 22 cm \varnothing and a length of 25 cm (FIG.6-08). It is simple to produce and has already been tested and measured by the Boelkow firm. A parabolic antenna (as in the case of Mariner) is unsuitable for thermal reasons.

FIG 6-07



Maximum of possible information flow for
a polydirectional and a directional antenna
as function of the flight period.

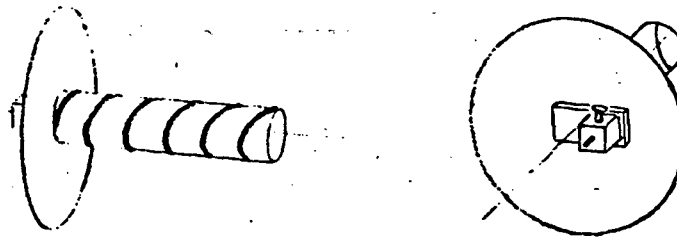
- a = information flow
- b = switch
- c = polydirectional antenna
- d = directional antenna
- e = flight period (days)

Tubes for the power amplifier are provided, which reach their highest efficiency and highest reliability in this frequency range. Especially favorable is the RH7Cc, developed by Siemens for Mariner Mars. It has very small dimensions and a weight of only 11 g. It is possible, without difficulty, to design the power amplifier redundantly. Its dimensions are then only 7.5 x 7.5 x 11 cm. For an efficiency of 40 per cent, the input capacity is 25 watt, which corresponds approximately to one third of the total capacity of the sonde.

The loss capacity of 15 watts is one of the most powerful heat sources within the sonde. For this reason, the power amplifier should be connected with the outer cell in a manner providing for good conductivity.

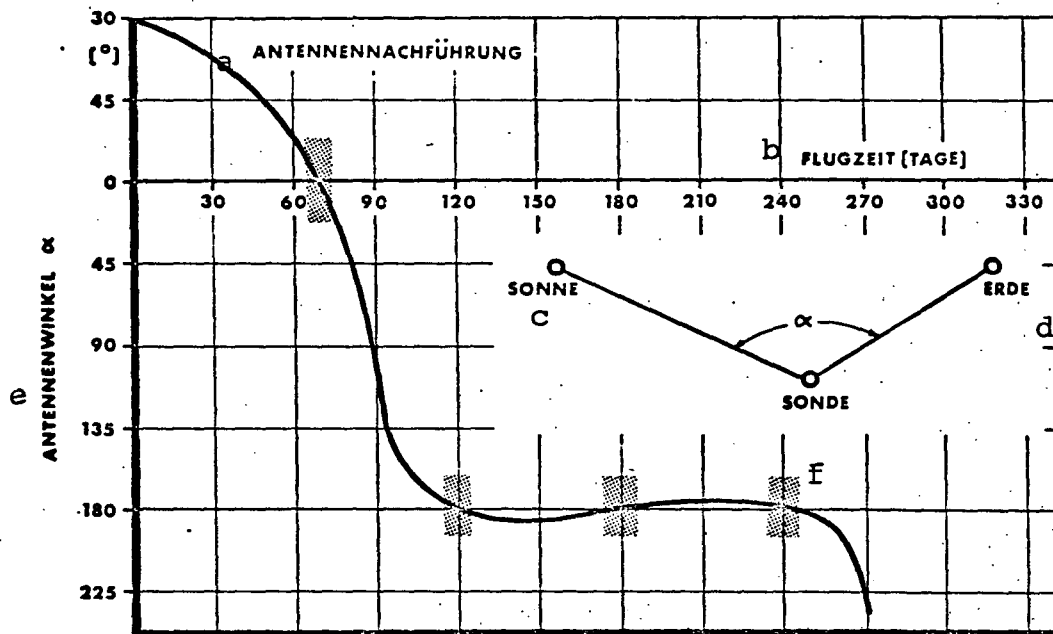
Alltogether, data transmission imposes no great demands upon technology. Pioneer VI and the Mars and Venus sondes of the USA have been successfully demonstrated in operation; there the reception antenna had an approximately 9 db smaller gain than the present parabolic 70 m antennas.

FIG 6-08



Special spiral antenna

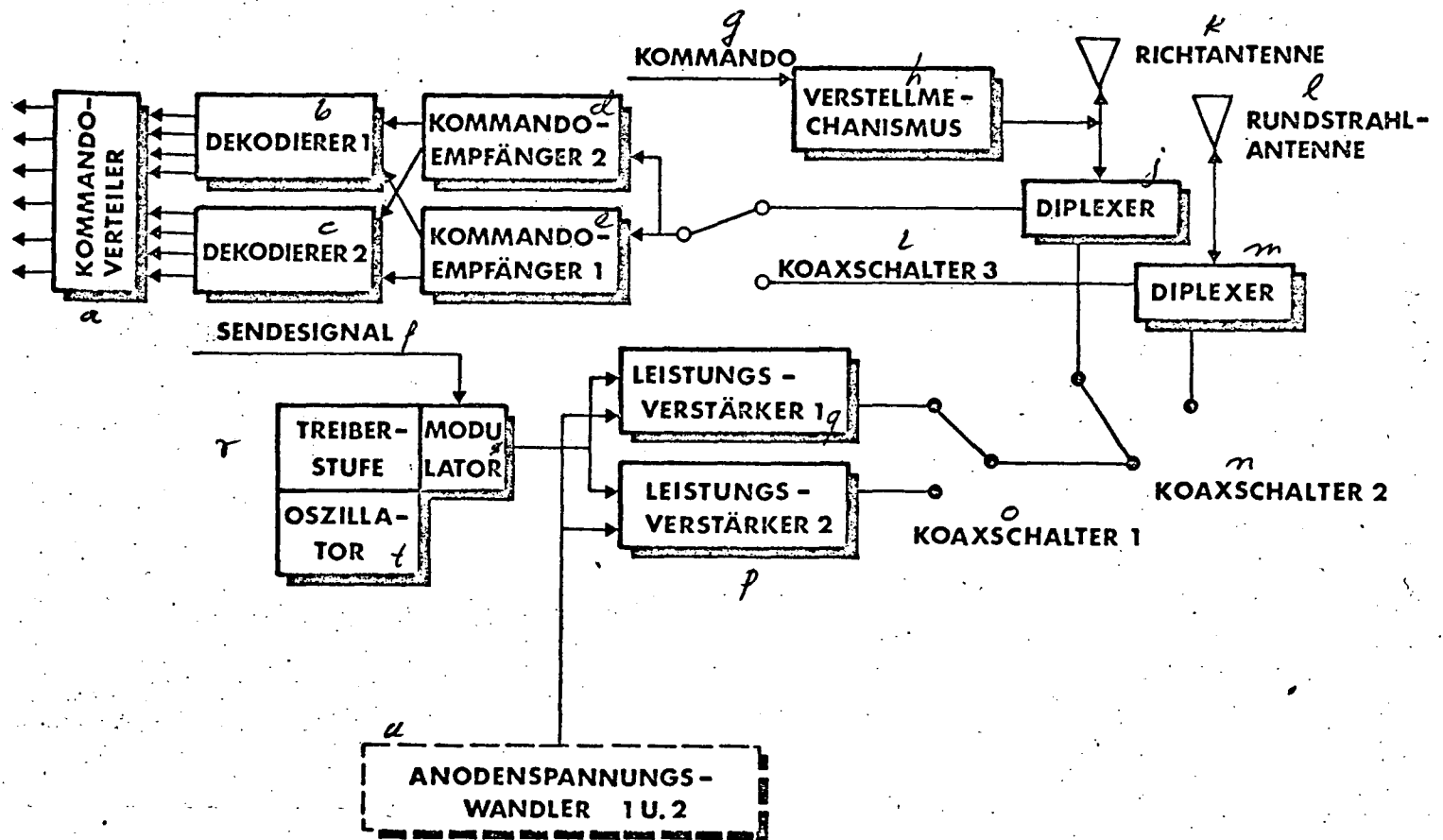
FIG 6-09



Antenna angle as function of flight time (shaded surfaces show the break-down periods during sun

passage). a= antenna trailing, b=flight period (days) c=sun, d=earth, e=antenna angle, f= sonde.

1 SCHEMA DER DATENÜBERTRAGUNG UND KOMMANDOEMPfang



1 = SCHEME OF DATA TRANSMISSION AND COMMAND RECEPTION

a= command distributor
 b= decoder 1
 c= " 2
 d= command receiver 2
 e= " " 1
 f= transmitter signal
 g= command
 h= adjusting mechanism
 i= coaxial switch
 j= duplexer
 k= directional antenna
 l= polydirectional antenna
 m= duplexer
 n= coaxial switch 2
 o= " " 1
 p= power amplifier 2
 q= " " 1

r= propelling stage
 s= modulator
 t= oscillator
 u= plate-voltage converter 1 & 2

6.3 Energy supply

The selection of the energy supply for flight missions near the sun is chiefly determined by three parameters:

- High radiation intensity
- High temperatures
- Particle-radiation load.

Batteries, radio nuclide generators and solar cells are in this case considered as energy bearers.

Batteries are out for flight periods exceeding 1 month, due to the high weight and the short lifespan. Moreover, the reliability is too low due to many possible fault sources in the case of long-period missions.

Radio nuclide generators can, in principle, be employed for a solar mission. However, due to the greater weight and the fact that, at present, safety regulations for flightworthy devices have not been precisely defined, the employment of solar cells would be the simplest and most economically justified possibility of energy generation. It will be shown in the following text that it is employable for a solar sonde.

For the total mission, a capacity of approximately 100 watt

is required. In order to secure this energy, arrangements of silicon cells of the n/p type were investigated, the reaction efficiency of which is 9- 11 per cent under vacuum conditions. Infrared and ultraviolet filters limit the main reaction area of the cells, in order to obtain an advantageous efficiency and to ward off heat rays.

The cell temperature is decisive for the design and capacity of the cell. FIGURE 6-10 shows the influence of the temperature. From this it can be seen that a temperature of approximately 400° K represents the load limit. This cell temperature depends upon the radiation intensity and thus upon the distance from the sun. In the case of direct insolation ($\varphi = 0^\circ$) a distance of ~ 0.66 AU is thus attainable.

A possibility of reducing intensity, and thus of influencing the equilibrium temperature, is offered by adjustment of the solar surfaces. FIGURE 6-12 shows this behavior with account taken of attainable α and ϵ values and the angle velocity. Here a change of the reflection characteristic of the cells at higher angles of attack is taken into consideration. It is shown that distances up to ~ 0.2 AU are attainable by adjusting the surface. Another parameter is the radiation load, which considerably influences the efficiency of the cell.

Influence of temperature upon the capacity of solar cells. a = capacity

FIG 6-11

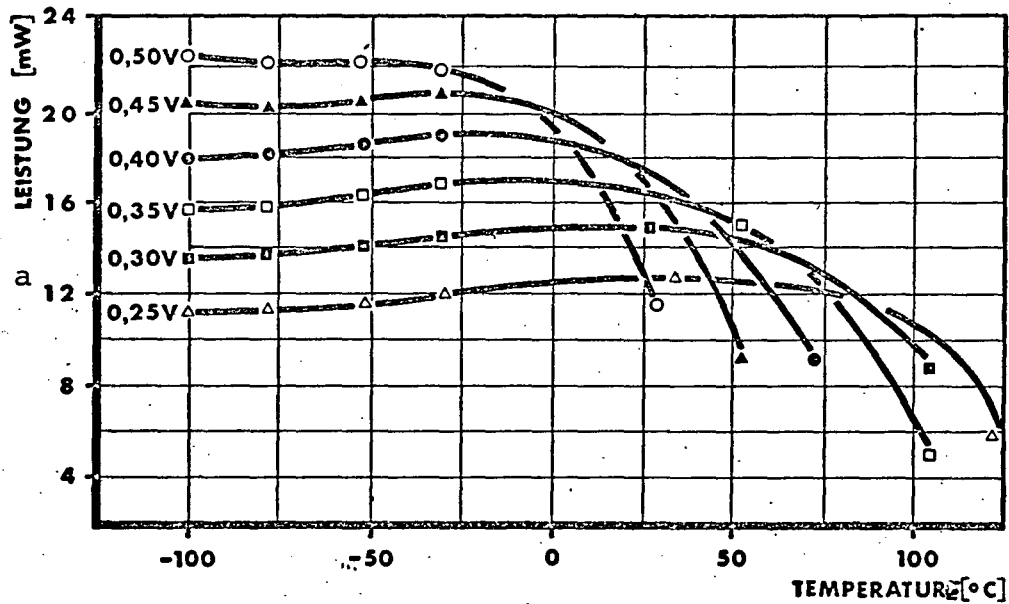
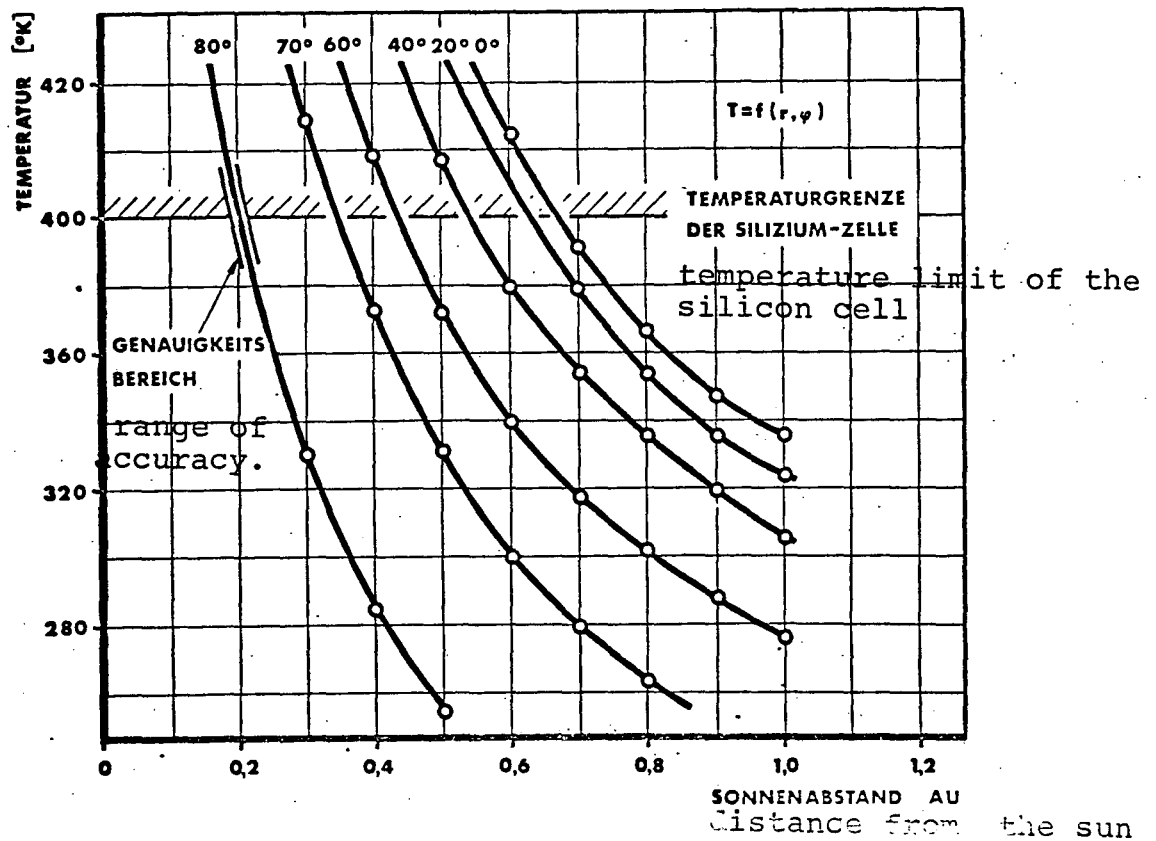


FIG 6-12



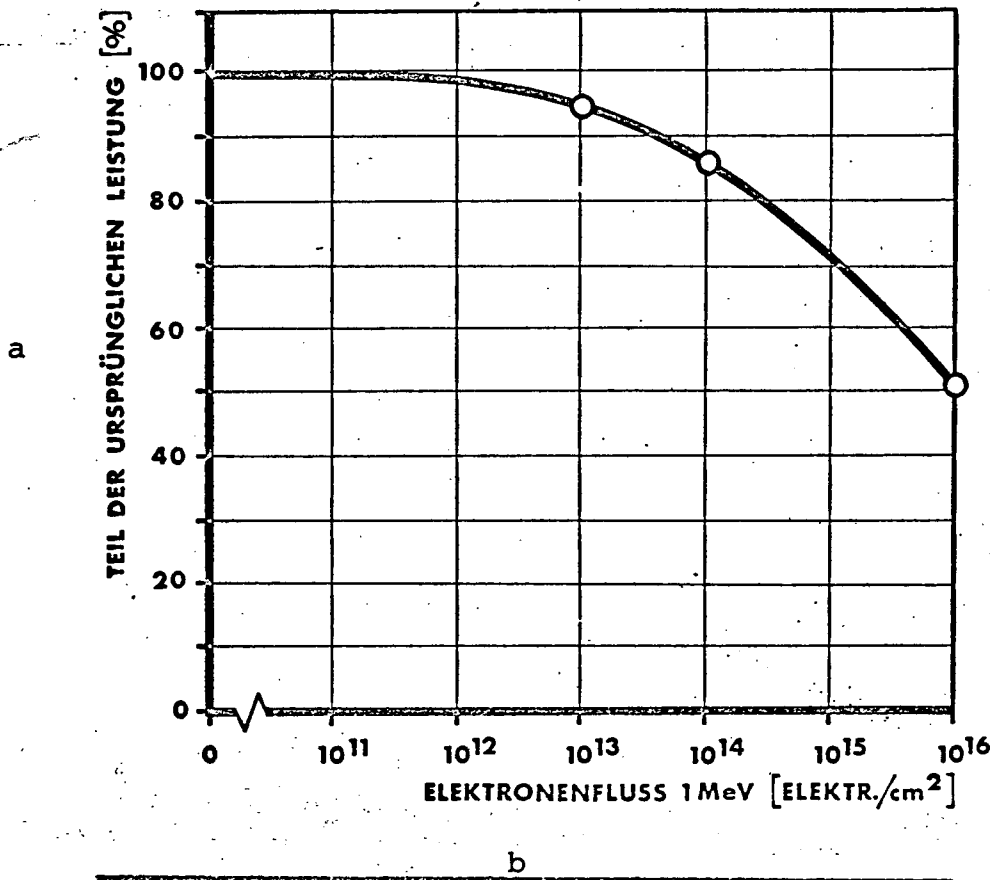
Temperature regulation by different angles of attack of solar cells.

Here this is only mentioned in general, since an estimate yielded no critical cases of stress. If higher loads should occur, a surface enlargement according to the drop of efficiency (FIGURE 6-13) must be reckoned with. The upper temperature limit decreases by approximately 10 - 15 ° C, so that then shorter distances are attainable.

Thus design proposition 1 would be an energy supply by means of 2 lateral solar surfaces of 0.5 m² each, which are pivoted around an axis perpendicularly to the sonde. There is here a choice between a minimum two-time switching and continuous switching (dotted line), to attain a required distance of 0.26 AU (FIGURE 6-14) and at the same time to ensure a 100 watt capacity for a minimum surface.

As an alternative to adjustable solar-cell surfaces, the new type arrangement shown in FIGURE 6-15 was investigated; the incident radiation is here reflected via a Fresnel mirror and a lens system to a cone, and from there to the laterally arranged solar surfaces. Here the adjustable lens system determines the required intensity on the cells. An estimate yielded the result that the temperature problems of the mirror and of the lens system can be mastered. The possibility of working in the optimal range of the cells offers an advantage.

FIG 6-13

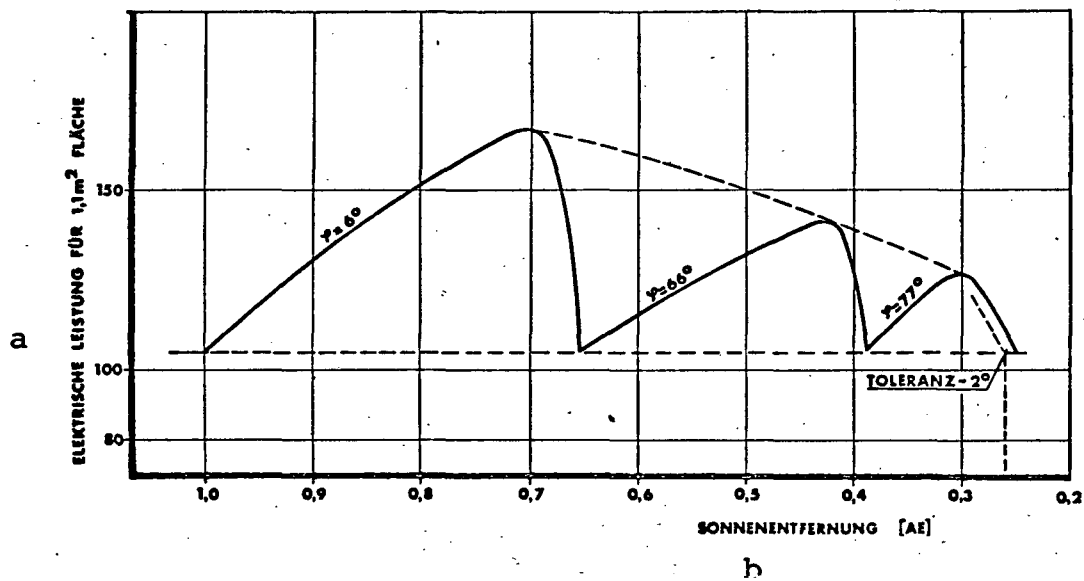


Decrease of efficiency by solar cells
through the influence of radiation

a = portion of the original capacity

b = Electron flow 1 Mev.

FIG 6-14



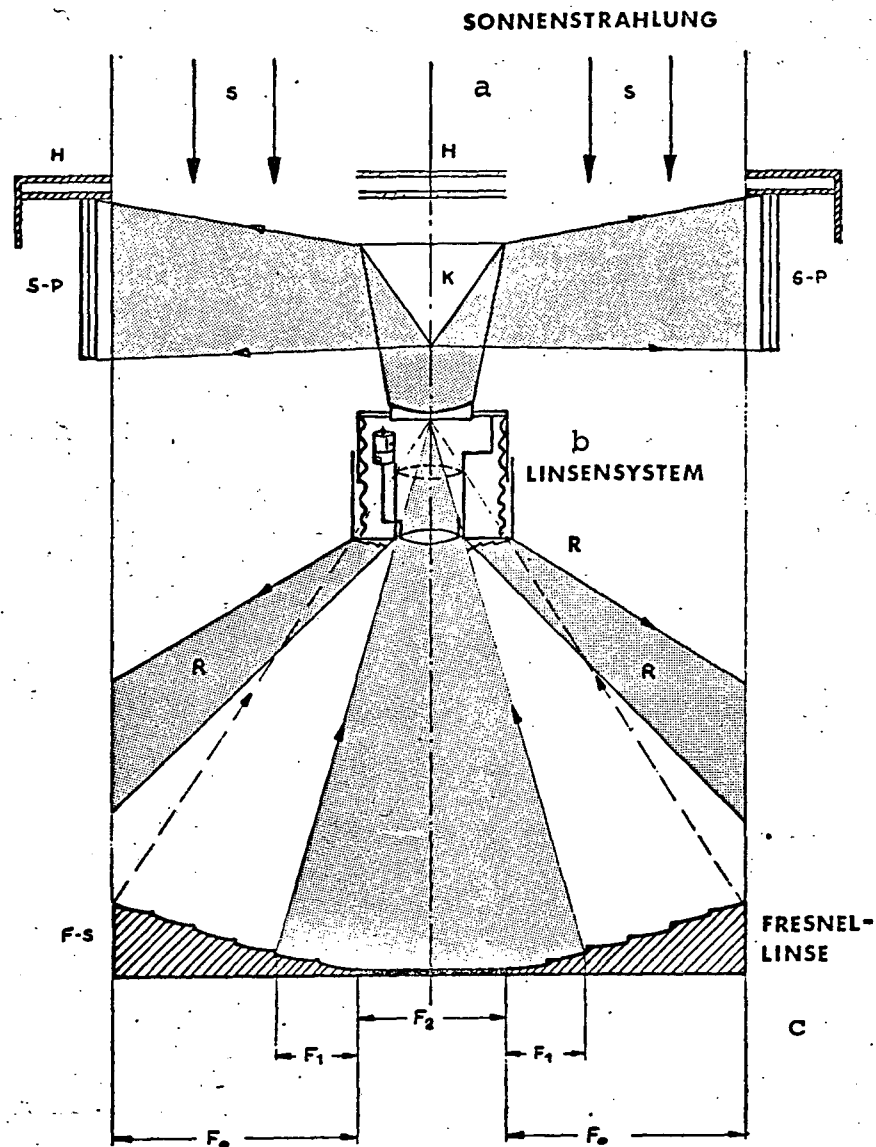
Power yield of the solar-cell surface
(1.1 m²) in the case of a three-
times switch of the angle of attack.

a- electric capacity for 1.1 m² surface

b- tolerance 2°

c- solar distance AU

FIG 6-15



Scheme of energy generation by means of a new-type system of lenses and a fixed solar-cell annular surface.

- a - solar radiation
- b - lens system
- c - Fresnel lens

The solar-cell surface is thus reduced, and the danger of damage to the cells by protons or electrons is excluded due to the perpendicular arrangement below the heat shield H.

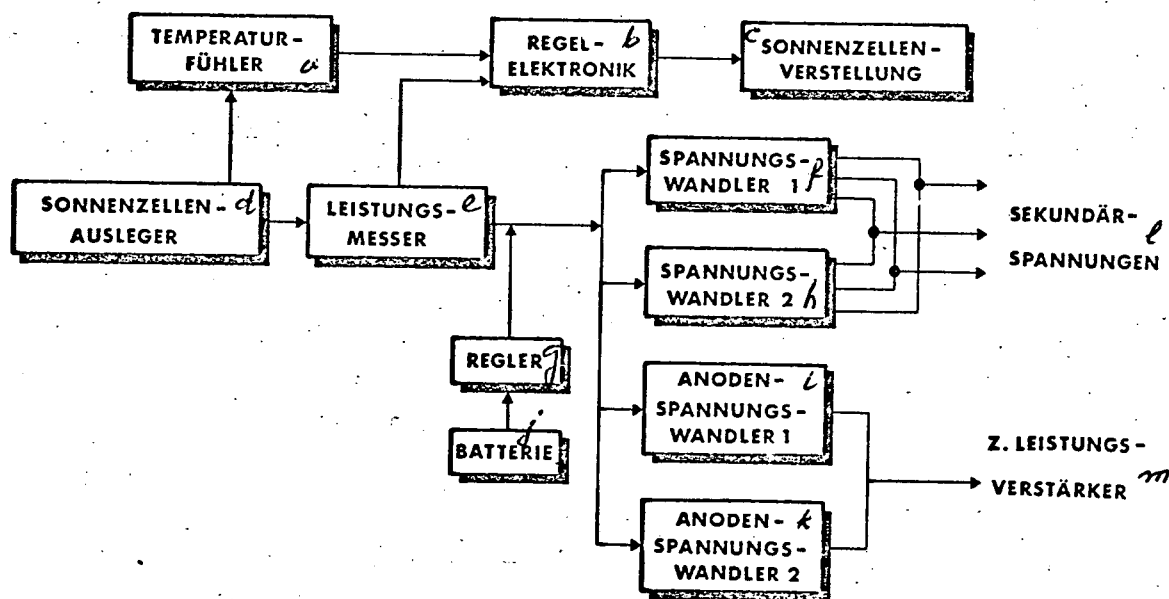
Since the adjustable lens system can select any intensity, an approach to the sun up to 0.1 AU would be thinkable.

The consideration thus show that the required capacity up to the requested distance of 0.26 AU is ensured by solar cells. (The more exact design parameters were investigated in TN-P6B-41/66).

Figure 6-16 shows the scheme of the complete energy-supply installation.

FIG
6-16

SCHEMA DER ENERGIEVERSORGUNG



1 - SCHEME OF ENERGY SUPPLY

- a - temperature feeler
- b - regulating electronic equipment
- c-- solar-cell adjustment
- d - solar-cell outrigger
- e - power meter
- f - voltage converter 1
- g - regulator
- h - voltage converter 2
- i - anode voltage converter 1
- j - battery
- k - anode voltage converter 2
- l - secondary voltages
- m - to power amplifier.

6.4 Heat Budget

6.41 Principles

In the case of an approach to 0.26 AU from the sun, the sonde gets into a region with a radiation intensity 16 times greater than in the vicinity of the earth.

This raises considerable problems of heat protection and temperature control, since no temperatures higher than 25 to 40° C are permissible for the measuring instruments and the electronic equipment.

The sonde temperature can, in principle, be limited by the following possibilities:

- (1) by interposition of one or several heat-protection shields
- (2) by expedient choice of the sonde configuration; and
- (3) by suitable selection of materials and surfaces for the critical planes.

Point (3) must be investigated first of all, in order to obtain the required fundamentals.

The ratio of solar absorptive power to the emissive power in the infrared range α/ϵ is decisive for the selection of a

heat-shield layer on the side facing the sun. The smaller is α/ϵ the lower will be the equilibrium temperature. Between shield and sonde the radiation takes place exclusively in the infrared range. In order to keep the passing heat flow low, the coverings of the surfaces standing here in the radiation exchange must have a small ϵ .

Some of the possible covering materials are listed in Table 6-IV.

TABLE 6-IV: Material Characteristics

Material	α/ϵ	ϵ	T_{\max}
White org. finishes	0,2-0,3	0,9	200 - 300° C
Black org. finishes	0,9	0,9	300 - 400° C
Anodized aluminum	0,12-0,3	0,9	350° C
Waterglass enamel	0,2-0,3	0,8	500° C depend. on base
Enamelled porcelain	0,25-0,35	0,8	600-700° C
Aluminum	0,22-0,4	0,6-0,8	dependent
Materials with low		ϵ	
Gold	0,02 - 0,05		
Silver	0,03 - 0,06		
Pure aluminum, polished	0,04 - 0,05		
Platinum	0,05 - 0,15		
Nickel	0,05 - 0,10		

TABLE 6-V¹ Influence of ultraviolet radiation upon α and ϵ .

Dye	Binder		before ²	after	before ϵ	after
ZnO	ZnO	K ₂ SiO ₃	0,17	0,33	0,95	0,93
Zircon	Zircon	K ₂ SiO ₃	0,13	0,42	0,95	0,93
Lithafrax	Lithafrax	K ₂ SiO ₃	0,12	0,32	0,93	0,89
SiO ₂	SiO ₂	Silicones	0,14	0,72	0,92	0,83
ZnO	ZnO	Methyl silic.	0,18	0,27	0,91	0,88

When selecting a covering material, it must be considered that α and ϵ change unfavorably through prolonged ultraviolet radiation as well as with increasing temperature. Here UV radiation has a particular influence on α . In Table 6-V can be seen change of α for some inorganic coverings after a 600-hour exposure to UV radiation of 10-fold solar intensity at a distance of 1 AU.

On the basis of the given values, these very well realizable values of α and ϵ were used for the following computation"

Heat shield, facing sun

$$\alpha = 0,3 - 0,5$$

$$\epsilon = 0,85$$

heat shield, facing sonde

$$\epsilon = 0,1$$

additional shields between sonde and front side of sonde

$$\epsilon = 0,025 - 0,05$$

6.42 Determination of Sonde Temperature for various shieldings

The investigated versions can be seen in FIGURES 6-17 and 6-18.

The following general assumptions were made:

(a) The internal heat of approximately 75 watt generated by the sonde is, for the most part, radiated off via a cylinder 0.6 m in diameter and 0.6 m in length (results from the position of the internal heat source).

(b) The heat transfer from heat shield to the sonde takes place exclusively by radiation.

(c) The material of the front side of the sonde conducts heat very well. The incident heat is conducted off via the front side of the sondes to the cylinder jacket, and is radiated off from there.

(d) With the exception of the cylinder surface considered for the radiation, all surfaces have a very low ϵ , so that the amount of heat radiated by these surfaces at identical temperature is almost negligible in comparison to the 75 watts.

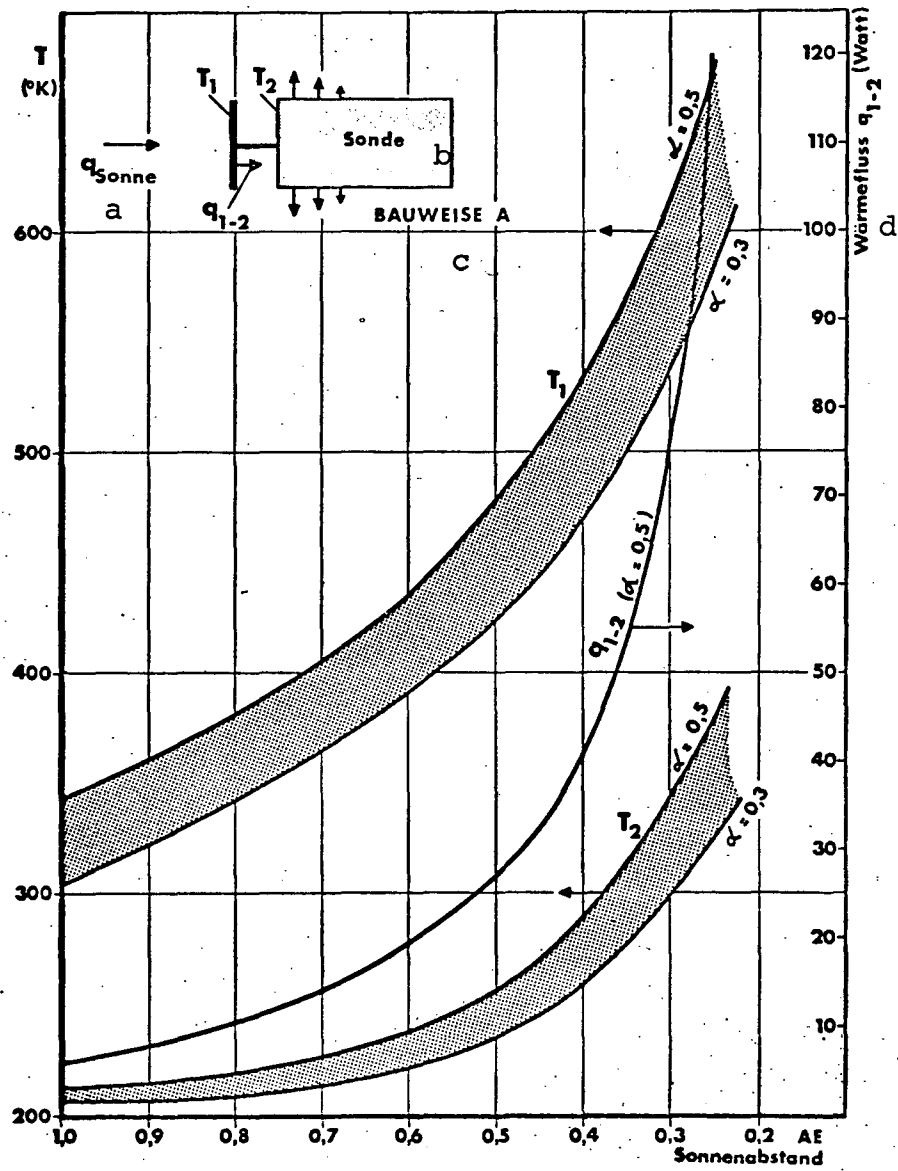


FIG 6-17

Heat flow and shield temperatures upon approach to the sun.

a=sun, b=sonde, c=type of construction
d=heat flow, e= distance from the sun

CONSTRUCTION TYPE A :

A heat shield with parallel front surface of sonde.

The configuration used as a basis for the calculation can be seen in FIGURE 6-17, the following special assumption being additionally made

- (a) the shield diameter is large in comparison to the shield distance
- (b) ϵ_{cyl} was so chosen that the radiation temperature

T_{∞} at a distance of approximately 0.5 m from the heat shield amounts to 200° C.

In FIGURE 6-17 the temperature change of the front surface of the sonde and the occurring heat-shield temperature as function of the distance from the sun are plotted for two values, $\alpha = 0,3$ and $\alpha = 0,5$. In addition, the heat flow conducted into the cylinder part is plotted.

Upon approaching the sun to 0,25 AU, a temperature increase for $\alpha = 0,3$ to 0,5 of 98° and 156° K is yielded on the front surface of the sonde. The corresponding maximum heat-shield temperature is approximately 585° K and 690° K. The temperature fluctuation on the front surface of the sonde is too high for the measurement instruments. It must, however, be taken into account that this higher temperature occurs only on

the front surface of the sonde, and very rapidly decreases to T_{∞} in the adjacent cylinder part, since a very high temperature gradient was computed at the transition point between the front and the cylinder part. It would therefore even be possible to keep this locally occurring higher temperature tolerable for measurement instruments and electronic by installing superinsulation foil within the sonde.

CONSTRUCTION TYPE B:

Two parallel heat shields with front surface of sonde parallel to them.

The following special assumption were made

- (a) the shield diameter is large in comparison the the shield distance
- (b) since in this case a smaller temperature fluctuation on the sonde must be reckonrf eiyh, the ϵ of the cylinder part was so chosen that at a greater distance from the front surface of the sonde, a temperature T_{∞} of 250° K occurs.

The computations show that the maximum temperature fluctuation on the sonde is only approximately 120° K. This value can still be considerably improved for $\alpha = 0.3$ or lower values of the ϵ on the sonde and intermediate shields.

The maximum occurring shield temperature is here 680° K. Here too, the temperature in the cylinder part adjacent to the front surface of the sonde decreases very rapidly, since a steep temperature gradient has resulted again at the point of transition between the front surface and the cylinder part.

The equilibrium temperature at the remaining sonde surface then amounts to approximately 250° C, in which case, however, the surfaces not included in the temperature control must have a very low ϵ .

CONSTRUCTION TYPE C:

Plane heat shield with conical front surface of sonde.

Configurations C and D (with two heat shields), shown in FIG. 6-18, yielded the most favorable values; for this reason that type of construction was chosen for the sonde.

The calculation of the curves shown in FIG. 6-18 were carried out under the following special assumptions:

(a) $T_{\infty} = 250^{\circ}$ K

- (b) In accordance with the complicated configuration, sight factors (F) had to be taken into account.

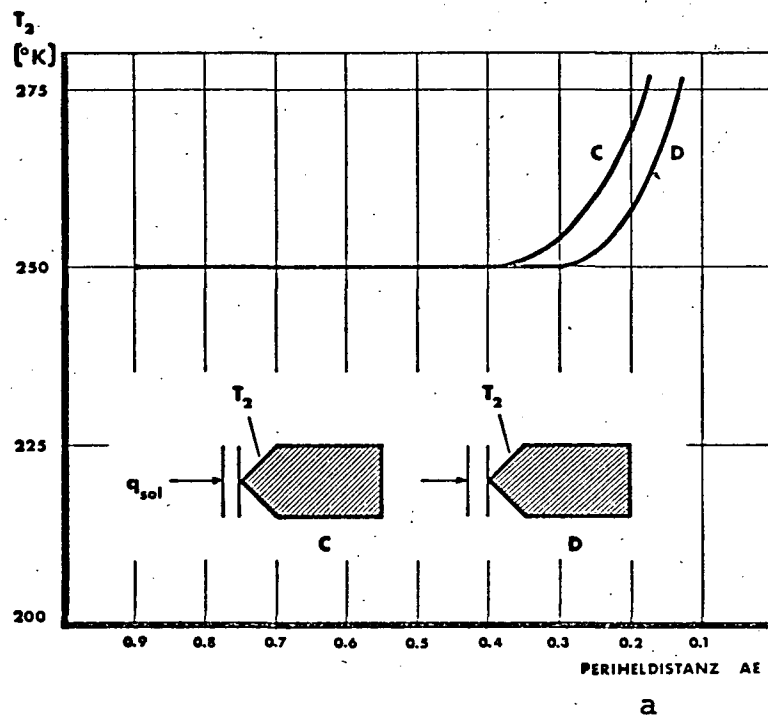


FIG 6-18

Rise of temperature on the front surface of the sonde with one and two heat shields.

a = perihelion distance, AU

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In accordance with the higher radiation, the cone temperature still rises by approximately 1.5° in the case of approach from 0.325 to 0.25 AU, and heat is conducted to the cylinder part. Conditions are similar in the case of configuration C; however, the temperature increase above T_{∞} already begins at 0.4 AU and is approximately 4° K at 0.25 AU.

It can be seen that by means of this sonde configuration and by suitable choice of finish, an extraordinarily good temperature constancy can be attained.

The three calculated examples show that no insoluble difficulties are to be expected with regard to the heat budget in the case of a solar sonde which shall approach the sun to within 0.25 AU. On the contrary, heat calculation shows still so many reserves that points of view of heat technology are not exclusively decisive in the choice of the sonde configuration, but the sonde configuration can to a large extent be determined by the instrumentation.

Only in the case of a still closer approach to the sun, is the sonde configuration to a considerable extent determined by thermal requirements, there being considerably still more favorable configurations than those shown in FIGURE 6-18.

6.5 Meteoroid and Radiation Protection

6.5.1 Meteoroid protection

Measurement instruments

the data-processing device

the telemetric device and transmitter unit, and

the position-regulation system with tank

must be protected against meteoroid penetration ;

antennas,

solar cells,

sensors, measurement feelers, etc.,

do not require special protection

For these parts, the possible number of impacts can only be kept at a minimum, namely by the smallest possible geometric aperture angles in the case of sensors, and as few as possible destructible cross-section surfaces for antennas and solar cells.

The following overlapping meteoroid environments can be defined:

- (a) zone near earth, approximately identical with the magnetosphere of the earth, up to approximately 100,000 km distance from the earth (FIGURE 6-19).

- (b) interplanetary space (at distances of more than 100,000 km from the earth, FIGURE 6-20).

After propellant cutoff of the kick stage, the sonde flies through zone (a) on a hyperbolic orbit. A period of 10 hrs is set as upper limit for the residence time. Meteoroid concentration in the interplanetary space outside of the activity sphere of the earth is less than one thousandst of the concentration near earth.

The frequency of meteoroids along the sonde orbit is on the average twice this value, i.e. , at the most 0.002 times the concentration in zone (a). This assumption represents a maximum.

In order to ensure a sufficiently long lifespan of the sonde, it is required that with a prediction probability of 99 % the above-mentioned systems will not be hit by meteoroids during the first half orbit. The transmitter unit should be protected against impact with a declared probability of 95 per cent during the first three years (6 orbits) if constants of celestial mechanics shall be determined by measurement of the sonde orbit.

Thus, the meteoroid protection must be designed for a life span of 10^h in the case of meteoroid flow densities according to FIG. 6-19 as well as for $0.25^a \approx 2500^h$ in the case of these



number

a= Kum. flow density(m² sec²)

b= rocket & satellite measurements (reduced)

c= theories

d= meteoroid mass

e= impact measurements

f= penetration measurements (reduced acc.to Bjork
 " " " " " Webb

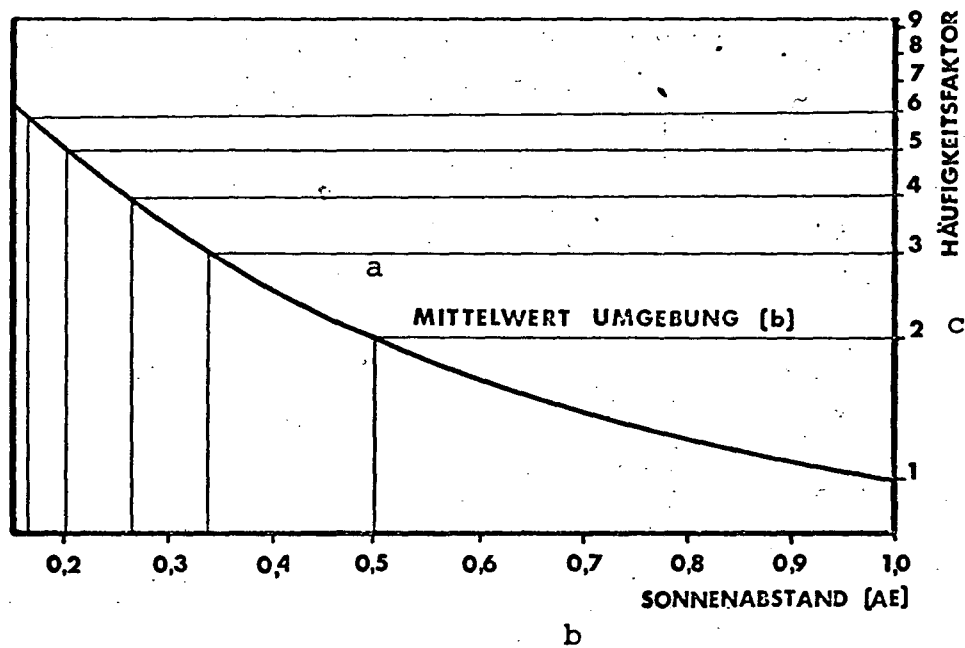


FIG 6-20

Expected increase of meteoroid density
near the sun (1 - 1/1000 of the density
near earth)

a= mean-value environment (b)

b= distance from sun (AU)

c= frequency factor

frequencies multiplied by 0.002. This corresponds to a required lifespan of 15^h in meteoroid environments (a).

The penetration probabilities as a function of the wall thickness are represented in FIGURE 6-21.

A lifespan of $3^a \approx 30,000^h$ is desired for the transmitter unit in zone (b). This, together with 10^h for zone (a) correspond to a life span of 70^h in environment (a); due to the small outer-wall surface of the transmitter unit, the penetration probability is yet considerably smaller than for the total system; special protection for the transmitter unit is not necessary.

Under the assumption that the systems to be protected are freely suspended in space without the outer skin of the sonde, and possess in each case individual outer-wall thicknesses of 0.5 mm (A1), a prediction probability of 93 per cent would result according to FIGURE 6-21 for 0 penetrations. Accordingly, an outer-wall thickness of 1.0 mm would be required for a safety of 99 per cent.

Since for constructional reasons the sonde is equipped with another outer skin, there is the possibility, proceeding from the above-mentioned system wall thicknesses (0.5 mm), to design the sonde skin as the "1st disk" of a two-disk meteoroid protector.

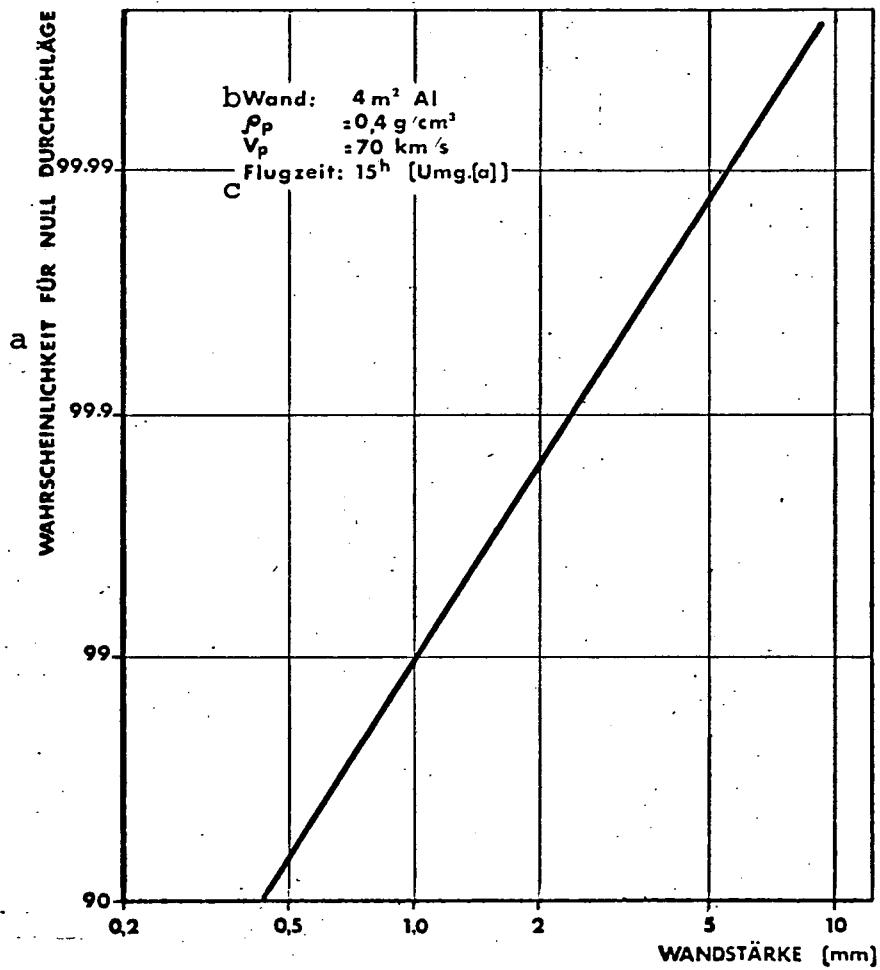


FIG 6-21

Probability of meteoroid impacts.

a- probability for zero penetrations

b= wall

c= flight period

d= wall thickness

Under the assumption of an outer-wall thickness of 0.5 mm (al) for a space between the two walls of 6 to 7 mm (without filling material between the walls), a safety of 99 per cent is assured. In this case additional meteoroid protection is thus not necessary.

More detailed information concerning meteoroid environments and meteoroid protection is contained in TN-P6B-43/66.

The ultimate design of the walls in a later phase of the project will have to take into account:

- a) the materials to be used
- b) the exact configuration of the sonde (distances between the walls),
- c) more accurate information, possibly available by that time, concerning meteoroid environment and impact mechanism.
- d) the actually required reliability of the individual systems,
- e) if occasion arises, possibilities of a tradeoff of the requirements.

6.52 Radiation protection

The highest cumulative particle-radiation load occurs during flight through the radiation belt of the earth. It will never be reached again, even in the case of proton flares and a required lifespan in the magnitude of years, through radiation from other sources (sun, galactic radiation). Experience gained hitherto with satellites and space sondes may therefore be used as a basis; special radiation protection is not necessary.

In detail, the following total flow densities are valid :

- a) electrons: cosmic radiation $\sim 3.2 \cdot 10^{-4} \text{ e/cm}^2 \text{ sec ster MeV}$
electron radiation belt $< 10^8 \text{ e/cm}^2 \text{ sec MeV}$

(Higher values are occasionally possible).

- b) Protons: cosmic radiation yields a total dose of $< 10^{-2} \text{ p/cm}^2 \text{ sec ster MeV}$ (maximum at approximately 100 MeV for a flow density of $10^{-1} \text{ p/cm}^2 \text{ sec ster in solar maximum}$)

The dose remains thus below $10^6 \text{ p/cm}^2 \text{ ster a.}$

During a proton flare the sonde receives a maximum dose of $10^6 \text{ p/cm}^2 \text{ ster}$ of $10^5 \text{ p/cm}^2 \text{ ster MeV.}$

The proton radiation belt of the earth contributes less than $10^8 \text{ p/cm}^2 \text{ sec MeV} \approx 10^{13} \text{ p/cm}^2 \text{ d MeV}$

The total dose in the energy range above 40 MeV is below $3 \times 10^4 \text{ p/cm}^2 \text{ sec} \approx 4 \times 10^3 \text{ p/cm}^2 \text{ d.}$

6.53 Surface erosion

Micrometeoroids of such small mass and velocity that they no longer penetrate the sonde cover, as well as energy-rich protons, may, due to surface erosion:

- a) change the reflection characteristics of the sonde and thus its heat budget;
- b) damage optically effective surface parts as well as solar cells.

Here too, experiences gained with the satellites and space sondes may be employed. In a later phase of the project, the effect of materials and outer wall heating upon the erosion resistance of the sonde will have to be investigated.

7. DESIGN OF PROJECT

7.1 Overall Concept of the Sonde

The configuration is chiefly determined by the position requirements of the experiments and their geometric dimensions. The experiment for measurement of the zodiacal light particularly influences the development of the configuration.

Eight measurements, evenly distributed over the circumference, must be conducted on a cone with an aperture angle of 30° and the longitudinal axis in the direction of the sun. Due to the weak intensity of the zodiacal light, the admission aperture must be large. At the same time, however, no sun light or scattered light may incide, i.e. the opening must be situated behind the heat shield.

The plasma experiment must be oriented with the main axis to the sun. Its field of vision, is however, narrowed by a slot. The analyser must be situated directly behind the heat shield.

The crude sensor must possess a dome-shaped field of vision, as unobstructed as possible, in order to orient the sonde after launching. The remaining electronic equipment is divided up into individual components and can be arranged almost arbitrarily. However, an adequate volume must be provided for.

The energy supply exert an additional influence upon the configuration. When a radionuclide generator is employed, the energy supply must, due to the high radiation dose, be fastened on an outrigger at least 1 m long; thus a considerable shielding weight must in addition be reckoned with.

If the energy supply is performed by solar cells, the great intensity, range between 1 and 0.25 AU can be accommodated by collapsing the solar-cell outrigger, or by varying the intensity by an optical system.

With these restrictions, the configurations A to F shown in Fig. 7-01 are in principle yielded. Variations D and E, which use a radio nuclide generator as source of energy, are for the time being eliminated for the following reasons

- under certain conditions, high weight due to necessary shielding
- high cost
- strict safety conditions.

However, radio nuclide generators become interesting if an approach to the sun of less than 0.2 AU is aimed at.

Configuration A is connected with a propulsion unit at the heat shield. It has a favorable center-of-gravity-position. Configuration B, which is fastened by the stern on the propulsion unit, is somewhat shorter.

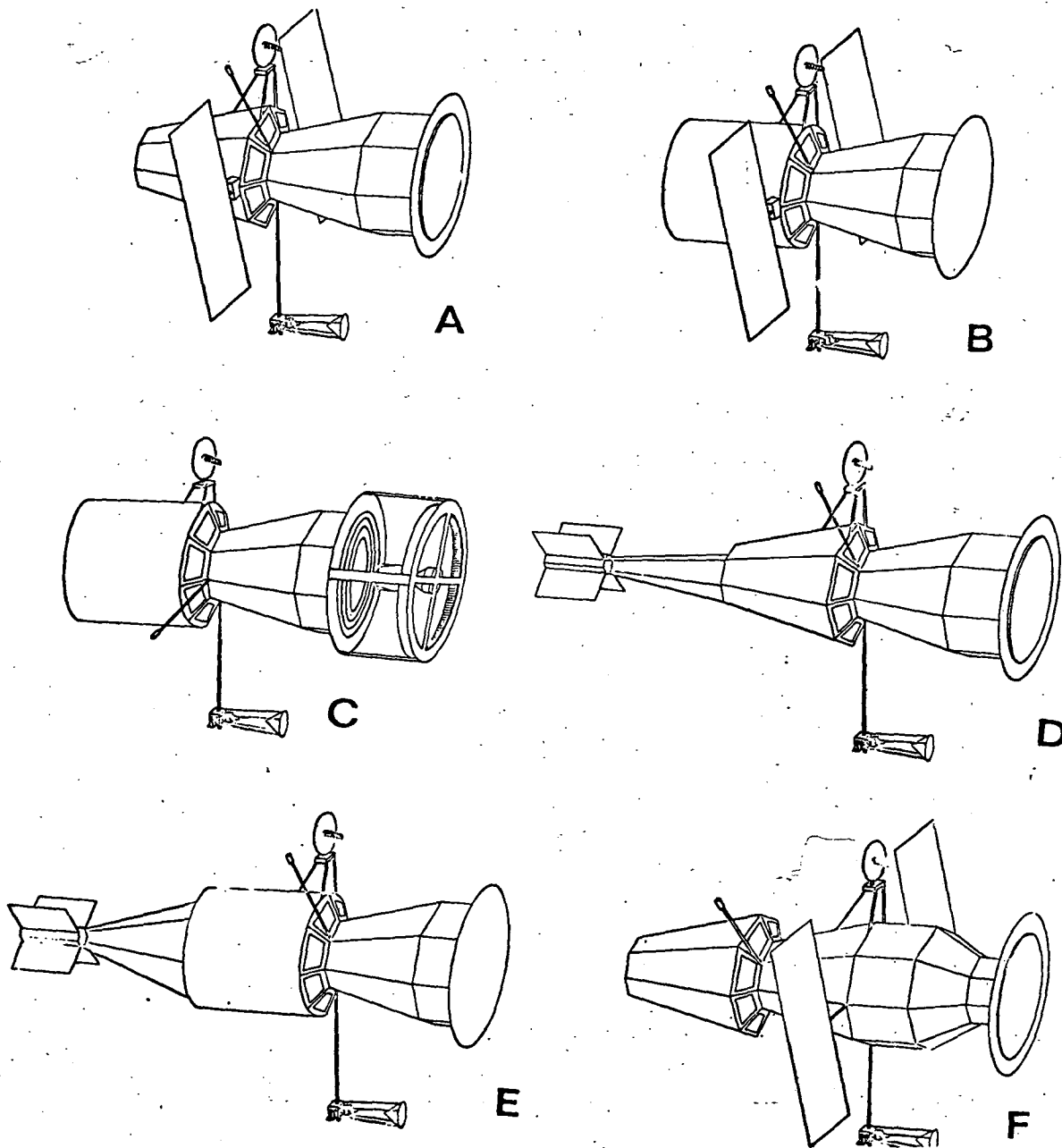


FIG 7-01

Survey of possible sonde configurations

Here the center-of-gravity position is no longer so favorable.

A variation with regard to energy supply is demonstrated by configuration C. Here, with the aid of a varioptic and a Fresnel mirror, a constant exposure of the solar cells is obtained, thus they can work in the optimal point. It seems, however, that this system is technically complicated.

Since the normal solar-cell outrigger are sufficient up to 0.25 AU even in this system, a configuration similar to A is chosen. Version F differs from A by a conic appendage. Thus a better heat constancy is achieved in the sonde. Moreover, the sonde can be fasted by a Marman ring of small diameter.

Sonde version F was worked out in a somewhat more detailed manner for the minimum useful load. In the following TABLE 7-I are compiled individual weights, dimensions, and the electric power consumption.

In accordance therewith, a total weight of 147 kg and a capacity requirement of 86.4 w is arrived at. In these values no redundancy, which seems to be necessary for reasons of reliability, is contained as yet. Above all, the command receiver, command decoder, capacity amplifier and the voltage converters should be installed in pairs.

The weight is not increased too greatly by this. But the reliability increases considerably, if the components are combined by cross-linking. First estimates show that approximately 60 to 65 per cent reliability, for a half-year flight period, can be reached without redundancy. By means of doubling the weakest components ($\Delta G = 10$ kg) an increase by approximately 10 per cent to approximately 70 to 75 per cent can be attained. A reliability of 80 per cent must be aimed at. In order to accomplish this, approximately 15 kg additional weight will have to be incorporated.

Exact and detailed investigations can be conducted only when the required program reliability and the carrier have been determined, and when the project is in an advanced stage.

However, it can already now be seen that in the proposed form, the sonde can be realized with sufficient reliability.

FIGURE 7-92 shows a three-side view of the sonde. For reasons of vibration engineering, almost all instruments must be mounted on a sandwich plate. With the exception of a few components, the arrangement can be made arbitrarily. Only the star sensor (14) is directed toward the celestial south pole. For reasons of thermal engineering, it is expedient to arrange the greatest power consumers (4,7,10,11) at certain fixed places.

TABLE 7-I: Preliminary weight balance of a solar sonde with 21 kg scientific instrumentation

	Weight	Measurement	Power	Remarks
1. MEASUREMENT DEVICES	g	cm	W	
1.1 Plasma analyser	10,0	20x20x30 8x 8x 8	5 5 only electronic	
1.2 Magnetometer	3,5			
1.3 Cosmic ray detector	2,0	30x20 ϕ	0,15	
1.4 Photometer(zodiacal light)	4,0	20x30 ϕ	2,0 only photometer	
1.5 Meteorite detector	1,0	20x10 ϕ	2,0	
	20,5		14,15	
2. DATA PROCESSING/STORAGE				
2.1 Analog/digital converter	0,8	5x10x15	0,3	
2.2 Digital-differential analyser	2,7	12x15x30	1,0	
2.3 Coder	1,0	10x15x12	0,5	
2.4 Tape storage	2,5	20x10x 5	2,0	10 ⁺
2.5 Nuclear storage	0,2	12x12x 1	0,2	
2.6 Address generator	0,5	5x 5x 8	0,4	
2.7 Command decoder	1,2	8x10x15	0,4	
2.8 Sequencer	2,5	12x12x20	2,0	
2.9 Computer	2,0	8x 8x10	1,5	
	3,4		8,3	
3. DATA TRANSMISSION/ COMMAND RECEPTION				
3.1 Modulator	0,3	5x 5x 8	0,8	
3.2 Transmitter	0,5	5x 8x15	0,7	
3.3 Power amplifier	5,0	10x10x30	25,0	
3.4 Synchronous generator and time giver	0,5	5x 8x10	0,5	
3.5 Command receiver	0,8	5x 8x15	1,0	
3.6 Polydirectional antenna	0,5	10x15 ϕ	-	
3.7 Directional antenna	4,0	22x22 ϕ	-	
3.8 Demodulator	-	-	-	Inc. in command
3.9 Duplexer and Coaxial switch	1,5	5x 5x 5	0,5 receiver	
	13,1		28,5	

+) peak capacity only for short time

	Weight	Measurement	Power	Remarks
	kg	cm	W	
4. POSITION REGULATION				
4.1 Sensors (without earth sensor)	3,8	10x14x35	4,0	
4.2 Position-reg. elec. equipment	3,0	20x30x10	4,0	
4.3 Position-regulation gas	0,1	-	-	
4.4 Gas tank -Tank	0,5	8,5 ϕ	-	
4.5 Pressure & temp. giver	0,2	5 x 2 ϕ	-	
4.6 Valves, pressure reducer, filter r	0,9	Σ 20x10x10	-	4 +)
4.7 Gas nozzles (12)	0,9	5x 4x 3	-	4 +)
4.8 Line system	0,2	-	-	
	<u>9,6</u>		<u>8,0</u>	

5t HEAT BUDGET/ALT /
TEMPERATURES REGULATION

5.1 Heat shield	10,0	-	-	
5.2 Blinds	2,0	-	-	
5.3 Temp. giver	0,5	-	0,1	
5.4 Regulating electronic equipment	0,5	-	0,5	
	<u>13,0</u>		<u>0,6</u>	

6. ENERGY SUPPLY AND SPACE BORNE
ELECTRICAL EQUIPMENT

6.1 Solar-cell outrigger	11,0	-	-	
6.2 Power regulator	1,0	10x 5x 5	2,0	
6.3 Voltage/converter	2,5	10x 8x 6	10,0	
6.4 Radionuclide generator	(25,0)	(30x60 ϕ)	-	
6.5 Puffer and auxiliary battery	1,5	10x15x 5	-	
6.6 Voltage transformer 25W/500 V	2,5	10x 8x 6	2,5	
	<u>18,5</u>		<u>14,5</u>	

7. STRUCTURE (Incl. METEORITE &
RADIATION PROTECTION)

7.1 Plugs and cables	5,0	-	-	
7.2 Upper cylinder portion	4,0	-	-	
7.3 Upper conus portion	5,0	-	-	
7.4 Middle cylinder with bottom	10,0	-	-	
7.5 Middle conus	4,0	-	-	
7.6 Lower conus	3,0	-	-	
7.7 Connection parts and arm	4,0	-	-	
7.8 Temperature & pressure giver	3,0	-	0,5	
7.9 Current & voltage meter	1,5	-	0,5	
	<u>39,5</u>		<u>1,0</u>	

-176a-

	<u>Weight</u>	<u>Power</u>
	kh	W
TOTAL	127.1	75.1
+ 15%		
safety	147.0	86.4

In the case of this light sonde, the center-of-gravity position is of subordinate importance. Nevertheless within the limits of possibility, care should be taken for a favorable position, i.e., if possible, a position on the longitudinal axis. The outriggers for the solar cells, the antennas, and the Foester sondes are fastened on the sandwich plate.

The total reaction forces is conducted via a cylinder or an octagonal prisma to a 45° cone. What crosssection shape the sonde will finally obtain depends upon dimensioning and the manufacturing process.

The cosmic-ray detector, which points 45° westward to the sun and lies within the ecliptic, is inserted into the transition cone. The force will finally be transmitted to the propulsion unit via a small cylinder and a Marman ring as connection element. The heat shields are fastened to the small cylinder. In the interior of this connection part are situated the plasma detector as well as the sun sensors. These structural parts are protected from extreme heat load by a three-fold shield.

The rear addition to the structure serves as housing for the mirror and its drive, which are needed for the zodiacal-light experiment. In the shadow of the last cone there is finally located the still very conveniently accessible meteorite detector.

FIG

7-02

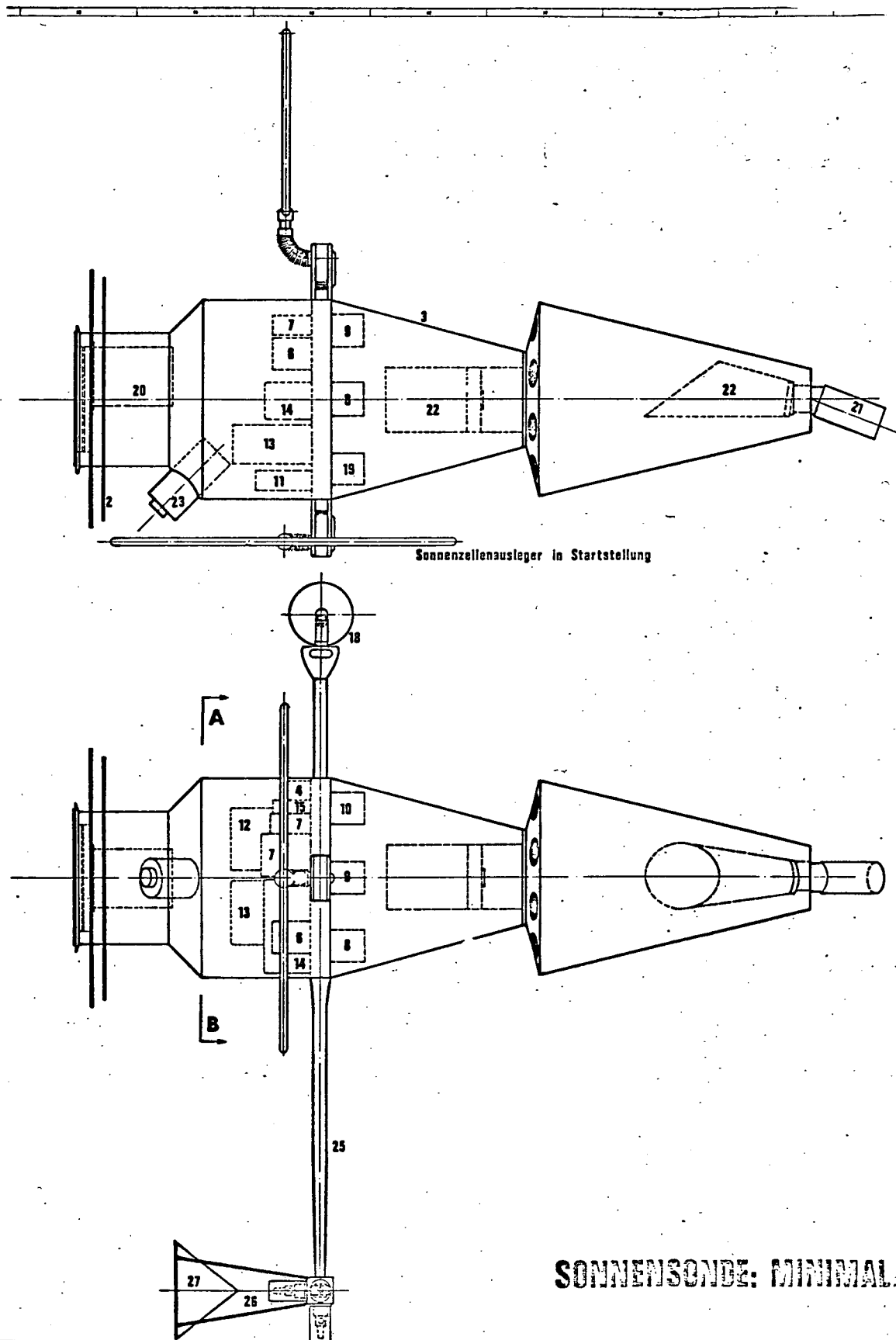
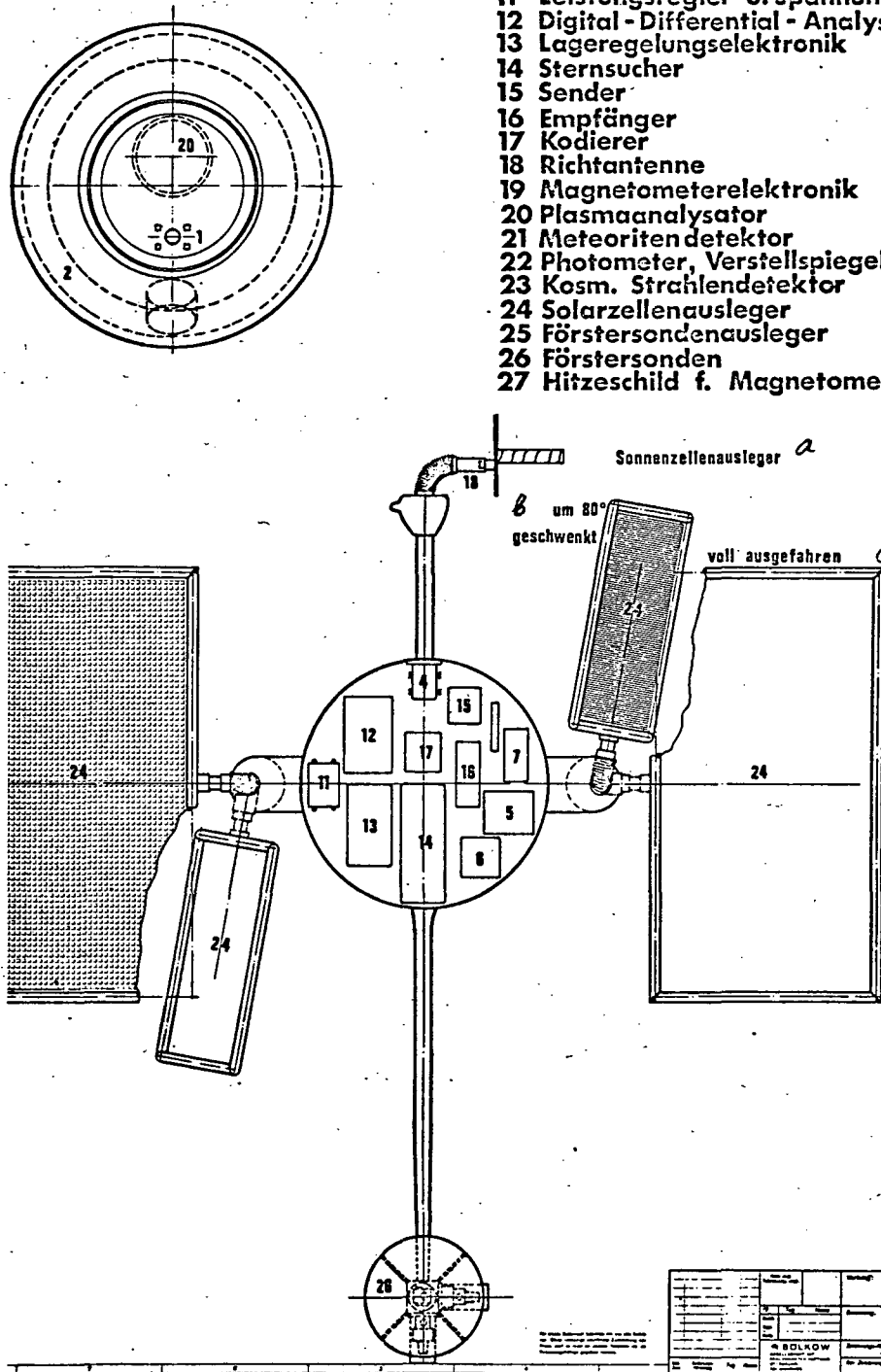


FIG. 7-02

(key to legend on the following page).

- 1 Sonnensensoren
- 2 Hitzeschild
- 3 Struktur
- 4 Leistungsverstärker
- 5 Sequenzer
- 6 Rechner
- 7 Bandspeicher, Kernspeicher
- 8 Zeitgeber
- 9 Kommando - Dekodierer
- 10 Batterie
- 11 Leistungsregler u. Spannungswandler
- 12 Digital - Differential - Analysator
- 13 Lageregelungselektronik
- 14 Sternsucher
- 15 Sender
- 16 Empfänger
- 17 Kodierer
- 18 Richtantenne
- 19 Magnetometerelektronik
- 20 Plasmaanalysator
- 21 Meteoriten detektor
- 22 Photometer, Verstellspiegel
- 23 Kosm. Strahlendetektor
- 24 Solarzellenausleger
- 25 Förstersondenausleger
- 26 Förstersonden
- 27 Hitzeschild f. Magnetometer

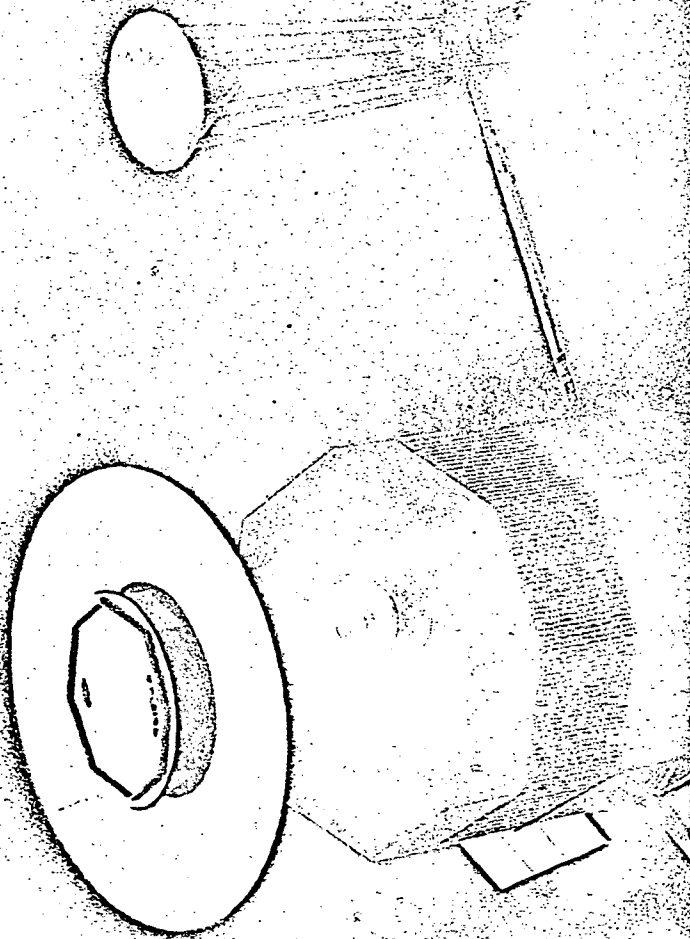


1. Sun sensors
2. heat shield
3. structure
4. power amplifier
5. sequencer
6. computer
7. tape

1. Sun sensors
2. heat shield
3. structure
4. power amplifier
5. sequencer
6. computer
7. tape storage, nuclear storage
8. time giver
9. command decoder
10. battery
11. capacity regulator and
voltage transformer
12. digital/differential analyser
13. position-regulation electronic equipment
14. star searcher
15. transmitter
16. receiver
17. coder
18. directional antenna
19. magnetometer electronic equipment
20. plasma analyser
21. meteorite detector
22. photometer, adjustable mirror
23. cosmic ray detector
24. solar-cell outrigger
25. Foester-sonde outrigger
26. Foester sondes
27. heat shield for magnetometer

a = solar-cell outrigger, b= pivoted by 80°, c= fully extended

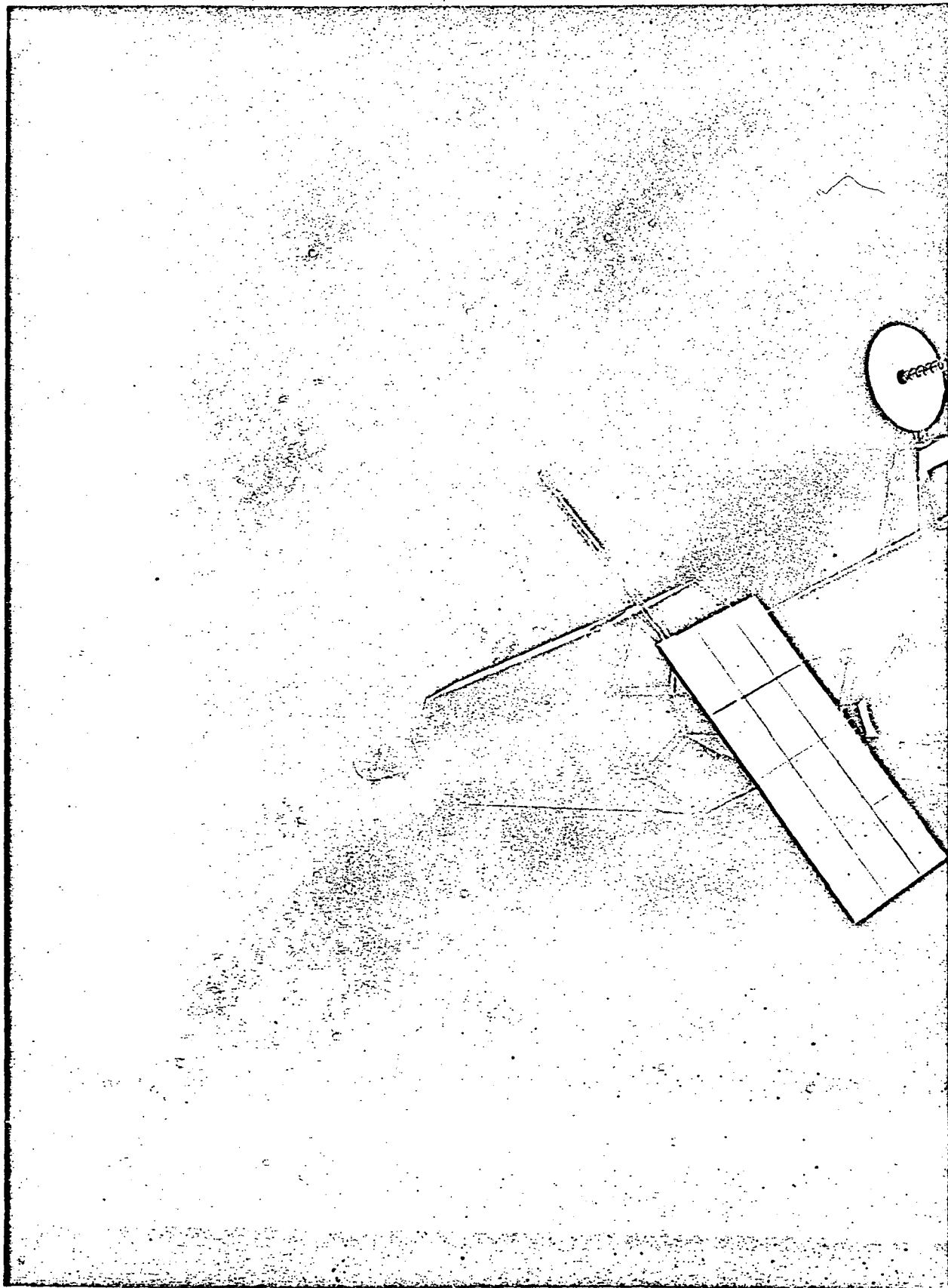
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FIG
7-03



The solar-cell outriggers as well as the directional antenna are pivoted by means of a vacuum crank. Thus it is possible to build a drive without bearing, which works in vacuum, in a closed atmosphere. There exists however, the disadvantage that with pivoting there is always connected a second movement. This disadvantage can however be eliminated for smaller pivoting angles (90°). However, for the time being drives with a vacuum crank were provided for and drawn.

The total concept is meant for the minimum version. It can, however, in principle also be used for the maximum useful load, since here only more measurement devices, to be sure with the same tasks and higher information flow, are installed.

The individual systems will be more extensive and will have to be designed more precisely. From the point of view of the picture, however, only the dimensions become greater, whereas the respective relationships are retained.

7.2 Plan of Development

Under the assumption that a project study will begin in the spring of 1967 lasting approximately 9 months, the earliest date to be reckoned with for the beginning of the development of the solar sonde is at the beginning of 1968.

The earliest date possible for the launching of a sonde under consideration of the propulsion unit is middle of 1972. If it is possible to shorten the development period for the propulsion modul (or if it is eliminated), the sonde could even be started at the beginning of 1972. Further reduction of the development period can only be purchased at the price of increased cost.

The building up of an appropriate organization will presumably take a year. This period of time depends to a considerable extent upon the intricacy of the organization scheme. The number of firms participating in the project is an important parameter.

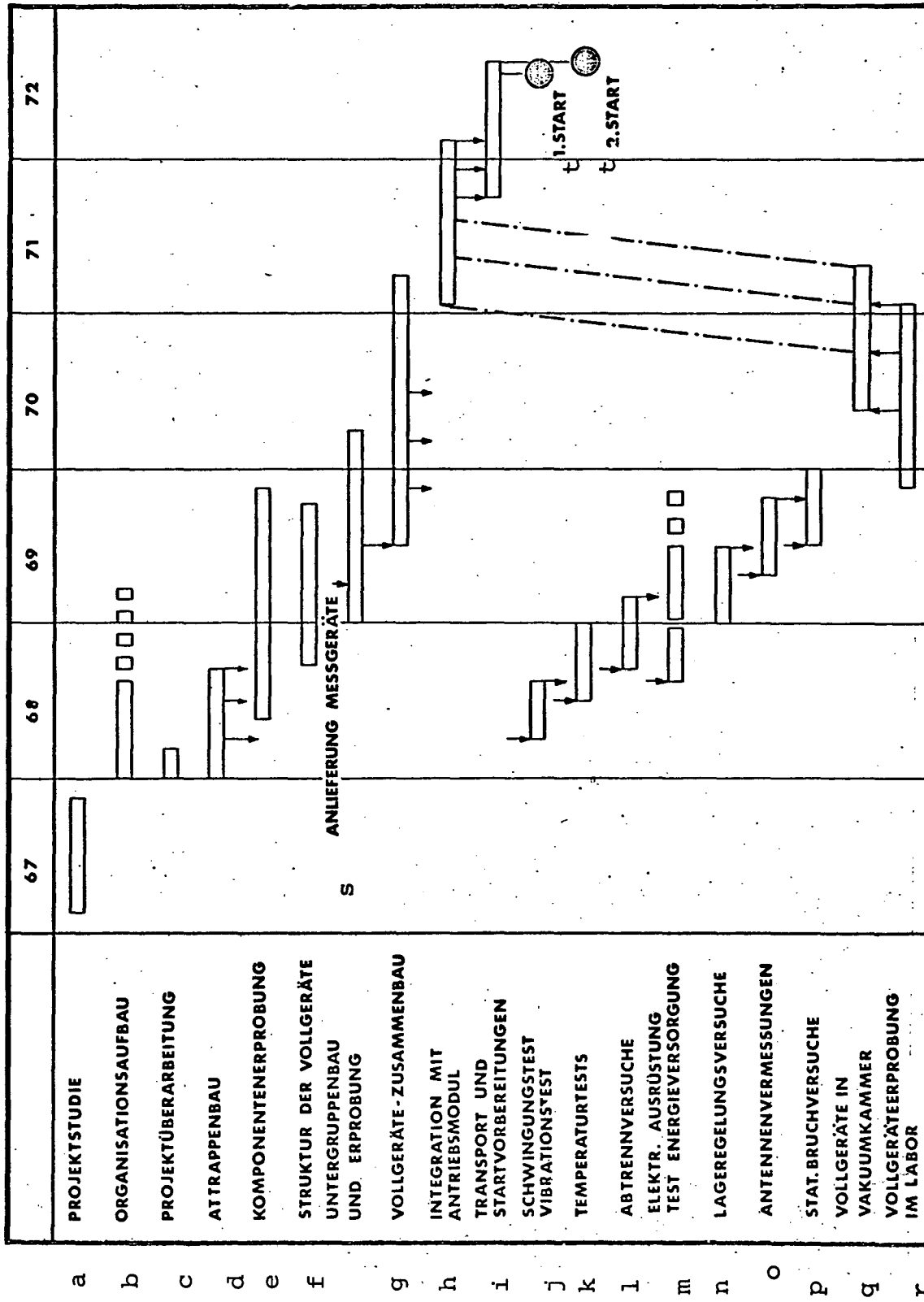
A development schedule, showing the most important events, is represented in FIGURE 7-04. It can be recognized that the individual intervals can in part be shifted among themselves or are exchangeable. There is thus a considerable safety margin that the devices are completed by the time of launching.

First of all, a brief phase of reworking of the project is scheduled. At the same time the construction of mockups is started. These simple devices are built up structurally like the complete devices. Only the equipment will, according to test, be simulated by simple dummies. Immediately after completion, the first oscillation and vibration tests will be conducted. With the second mockup, investigations on the heat shield and the covering

of the outer cell will be conducted in a high-vacuum chamber with sun simulation.

The third mockup is used for separation tests. After the first test series, the devices are re-equipped and the following experiments are conducted:

- energy supply and electric wiring,
- position-regulation tests, and
- antenna measurements.



(key following page)

- a PROJECT STUDY
- b ORGANIZATIONAL BUILD-UP
- c MOCKUP CONSTRUCTION
- d COMPONENT TEST
- e STRUCTURE OF COMPLETE DEVICE
- f SUBASSEMBLY TESTING AND CONSTRUCTION
- g ASSEMBLY OF COMPLETE DEVICE
- h INTEGRATION WITH PROPULSION UNIT
- i TRANSPORT AND LAUNCHING PREPARATIONS
- j OSCILLATION TEST
VIBRATION TEST
- k TEMPERATURE TESTS
- l SEPARATION EXPERIMENTS
- m ELECTRIC EQUIPMENT
TEST ENERGY SUPPLY
- n POSITION-REGULATION EXPERIMENTS
- o ANTENNA MEASUREMENTS
- p STATIC FRACTURE TESTS
- q ACTUAL DEVICES IN VACUUM CHAMBER
- r TESTING OF ACTUAL DEVICE IN THE LABORATORY
- s DELIVERY OF MEASUREMENT DEVICES
- t 1st LAUNCHING
- t 2nd LAUNCHING

After completion of this series two of the mockups can be used for static fracture tests or other purposes. The third one continues to serve as an installation device in the case of possible changes.

Component tests, which are continuously expended to construction division, are conducted simultaneously with these investigations which concern the total device. At the end of the test then rank the examinations of entire subsystems, which can be installed in the actual devices.

The fully equipped sondes are then thoroughly tested in laboratories and on test stands prior to being readied for integration experiments with the propulsion unit. After the successful completion of these tests, the sonde may then be moved to the launching site where the launching can take place after a preparation period of approximately 1/2 year.

Altogether, 3 mockups and 4 complete devices are required for the development program:

Mocup 1:

This mockup is started immediately after the project study. It is employed for:

- oscillation tests
- vibration tests
- experiments with the energy supply
- wiring (cabling) investigations

Mockup 2:

With this mockup the following investigations are conducted:

- temperature tests
- position-regulation tests
- fracture tests

Mockup 3:

- separation tests
- antenna measurement
- fracture tests

Complete device 1 and 2"

Two complete devices are needed for the launching. The production of the devices and of the structure is started approximately one year after placement of the order. There remains sufficient time for the equipment, integration with the propulsion unit and the transport to the launching site.

Complete device 3:

This device is used as back up. It is used in the case of

breakdown of one of the two complete devices.

Complete device 4:

Development tests device in flight construction. For the manufacture and tests of components, construction groups and devices

- dustfree labs
- a dust-free assembly hall
- small vacuum chambers
- one large vacuum chamber with sun simulation
- an antenna measurement place
- a test stand for separation experiments and
- a 3-axis test stand

are required.

The largest part of these installations exists or is being constructed for the development of other projects (for example 625 A 1).

7.3 Estimate of Development Cost

7.31 Introduction

At the present time there does not yet exist a reliable method for the determination of the development cost of space projects. There are only two possibilities:

- (A) Statistical evaluation of the cost of development of American devices, and
- (B) Determination of the cost of development by careful analysis of all systems.

The latter method presumes however, that there exists a concrete technically defined project and an organization for development. Since this does not yet hold true in the case of this feasibility study, there remains only the possibility of presenting a survey of the most important influence factors with the aid of the statistical evaluation of American experiences.

The goal is here:

- (1) Determination of the order of magnitude of the cost of development of a solar sonde, and
- (2) Representation of possibilities of reduction of the cost of development.

7.32 Cost of development of similar devices

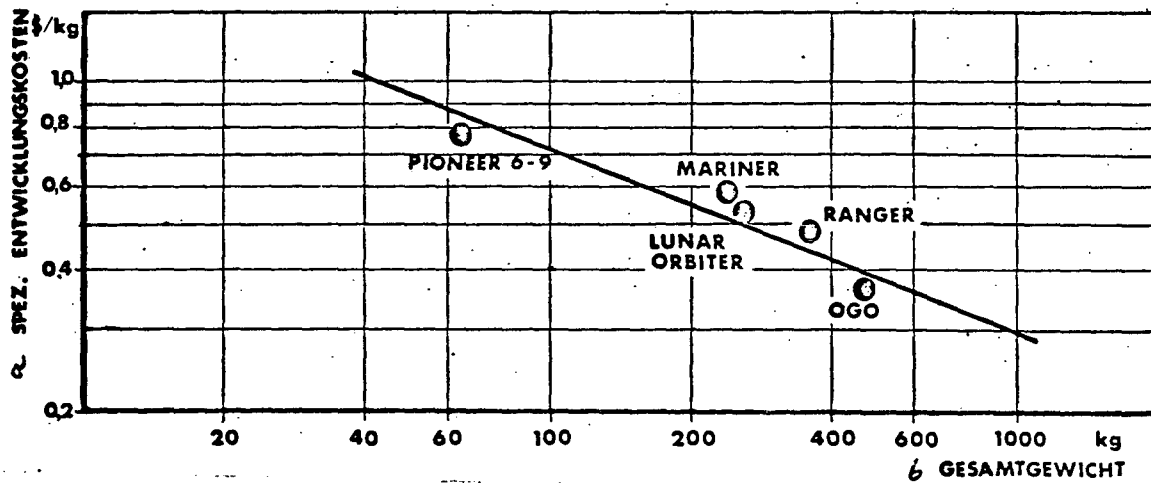
In the USA the cost of development accumulated within the scope of progressive satellites and space-flight devices can be determined by checking the NASA budgets of the past years. Only these figures may be considered reliable, since in most cases the original estimates of cost as well as the first contract sums do not hold true.

TABLE 7-II shows the result of this compilation; the evaluation with regard to the specific cost of development is represented in FIGURE 7.05.

RANGER	13	358 kg	47	13%	170	0,48
MARINER I - IV	7	235 kg	23	10%	137	0,58
OGO	6	384 kg o. Exp.	86	18%	136,9	0,36
PIONEER 6 ff.	6	48 kg o. Exp. a	16	25%	37,4	0,78
LUNAR ORBITER	7	250 kg o. Treib- stoff b	ca. 70	28%	130	0,52

a=without exper.
b= " fuel

FIG



Specific cost of development of some American space-flight devices and larger satellites.

a= specific cost of development , b= total weight

FIGURE 7-06 a shows an evaluation of this curve. It would yield approximately 80 million dollars cost of development for a solar sonde with 125 kg net weight. This figure, according to TABLE 7-II, refers, however, to 6 to 7 devices. If the construction number is reduced to 4 devices (2 flight devices, 1 back up, 1 test device), approximately 72.5 million \$ cost of development is yielded. This value must, however, be considered an average value; it is strongly influenced by the question of the extent to which technology can be stressed in favor of weight.

If the share of scientific instrumentation with regard to the total weight (in per cent) is introduced as measure, the curve of FIGURE 7-06 b is approximately yielded. If the statements of the first two diagrams are combined, an optimal construction weight is surprisingly yielded for the sonde (FIGURE 7-06 c), which requires a minimum of cost of development (approximately 120 kg to 140 kg mass without instrumentation). The cost are then in the vicinity of approximately 57 million.

If it is desired to increase the reliability by redundancy (and thus increased weight), the expenditure for development will of course grow as well. FIGURE 7-06 shows the expected rates of increase.

A cost formula independent of this method by ITT Research Institute, Chicago (Report No. C-7, "Spacecraft Program Cost

Estimating Manual") yields for the solar sonde, defined in this report with 147 total weight and 21 kg scientific instrumentation, an amount of

fifty-four million \$

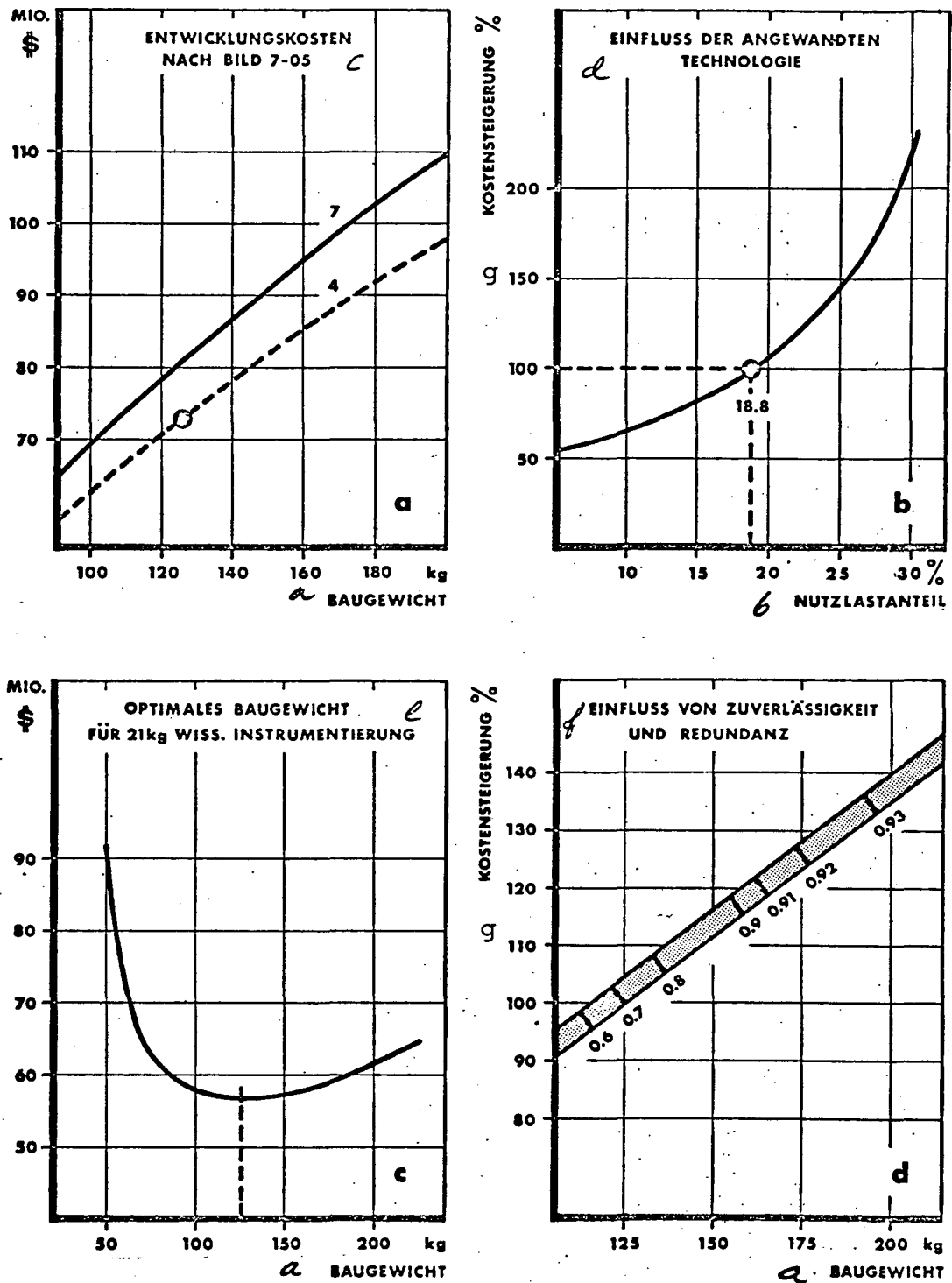
as cost of development, thus very good agreement.

It remains to discuss, as final factors, the influences of the project organization and of the conversion of US values to DM.

- (1) Experiences of recent years have shown that the project costs can easily be doubled, unless an optimal organization with brief, direct ways and a fully responsible chief contractor is chosen. This factor must by no means be disregarded.
- (2) The conversion rate \$/DM is officially at 4.0 and, according to general comparisons of standard of living, is on the average 2.6. The correct conversion factor for the cost of development is likely between these values.

From this is yielded the fact that the cost of development of the solar sonde may be in the optimal case 140 million DM, but may increase, with poor

FIG 7-06



Influence of the construction weight and number of devices (a), of the technical standard (b), and of the reliability upon the cost of development of a solar sonde with 21 kg scientific instrumentation.

a=construction weight, b=share of useful load, c=cost of development according to Fig.7-05, d=influence of applied technology, e=optimal construction weight for 21 kg scient. instrum. f=influence of reliability of redundance, g=cost increase.

organization and other unfavorable influence, to more than 300 million DM.

It shall be a main goal of the following project study, to show ways and possibilities of the development at minimum cost.

7.4 Project Study

As a next step after this feasibility study and prior to the beginning of the actual development, a detailed project study is required.

This project study will use, as specifications, the present work results as well as additional information or restrictions.

Its duration will be approximately 9 months and the cost is approximately 0.0 to 1.5 million DM.

The main goal of the project study is the definition of concrete statements concerning the technical data of the project, the development period and the necessary cost.

The project study should start as soon as possible, since each delay has direct influence upon the development and thus the date of launching.

8. SUMMARY OF THE RESEARCH RESULT

The study has shown that it will be possible to develop without special technical difficulties, a solar sonde for approximately 0.26 AU perihelion with a high degree of probability of success.

For launching of the sonde, either the SATURN V carrier rocket can be used

- a launching with two sondes which can have extended instrumentation, or the carrier rocket ATLAS-CENTAUR

- two to three launching with an additional kick stage or a propulsion unit, which would possibly also have to (and could be) developed in Germany.

The weight of the sonde with minimum instrumentation of 21 kg amounts to approximately 150. to 165 kg; the cost of development are above 140 million DM.

The cost of development for the highly energetic propulsion unit with 1.6 t fuel is estimated at 195 million DM. This propulsion unit can, however, also be employed for other carrier rockets and for other interplanetary missions.

9. CONCLUSION

In accordance with the study contract, Enclosure A, III.1, the following findings are made:

- (1) American data available as basis for the separation of this study were only those with regard to capacity analysis and orbit mechanics, which, however, had to be revised. No corresponding research is known which led to the technical design of a solar sonde (with the exception of a nuclear-electric sonde for SATURN V).
- (2) No American consultation was considered with regard to the preparation of this study.
- (3) No patent rights were knowingly employed in this research, and none have arisen.
- (4) Individually conducted works, which led to this research report, were laid down in "Technical Notes". A list is attached as appendix.

APPENDIX:

List of Reports and Technical Notes Written Within the Scope of this Study.

- (1) TN-P6B-31/66 W. Kokott
Problems of the scientific Propositions and useful-load integration
in an ecliptical solar sonde
- (2) TN-P6B-32/66 D. E. Koelle
Possibilities of the injection of a solar sonde into an orbit with
0.20 to 0.40 AU perihelion
- (3) TN-P6B-33/66 H. Schweig
Estimate of the weight of a solar sonde for 0.3 AU distance
from the sun
- (4) TN-P6B-34/66 H. Popp
Scientific instrumentation of interplanetary space sondes
- (5) TN-P6B-35/66 H. Schweig
On the mission reliability of a solar sonde
- (6) TN-P6B-36/66 H. Schweig
The net weight of a highly energetic propulsion unit for solar
sondes
- (7) TN-P6B-38/66 H. Schweig
On the position regulation of a solar sonde
- (8) TN-P6B-40/66 W. Kokott
Influence of the scientific measurement program upon telemetry
and position regulation of an ecliptical solar sonde
- (9) TN-P6B-40/66 H. Heppeshausen
Investigation concerning the employment of solar cells for the
energy supply of missions near the sun.

- (11) TN-P6B-42/66 H. Schweig
On the position stabilization of a solar sonde
- (12) TN-P6B-43/66 W. Kokott
Meteoroid protection for an ecliptical solar sonde
- (13) TN-P6B-44/66 H. Schweig
On the data transmission of the solar sonde in minimum version
- (14) TN/P6B-45/66 H. Popp
Installation conditions for the scientific experiments of an
ecliptical solar sonde
- (15) TN-P6B-46/66 P. Kleber
Nuclear-electric propulsion unit for a solar sonde
- (16) TN-P6B-77/66 W. Kokott
Tape width and possible information flow for the data transmission
of an ecliptical solar sonde
- (17) TN-P6B-48/66 M. Leupold
Design of a propulsion unit for the injection of a solar sonde
- (18) TN-P7D-3/66 Dr. Hennen
Principles of the thermal radioisotope propulsion
- (19) FM-355 R. Metzger
Orbits of solar sondes making use of strong strong perturbations
by the planets Venus and Jupiter
- (20) TN-P6B-23/66 K. Bohnhoff
On the design of a highly energetic kick stage for interplanetary
missions.
- (21) FM-345 R. Metzger
The geocentric ascent phase of interplanetary transition orbits
- (22) TN-P6B-49/66 K.-O. Sippel
Thermal problems of a solar sonde